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The Journal of
THE ROYAL AERONAUTICAL SOCIETY

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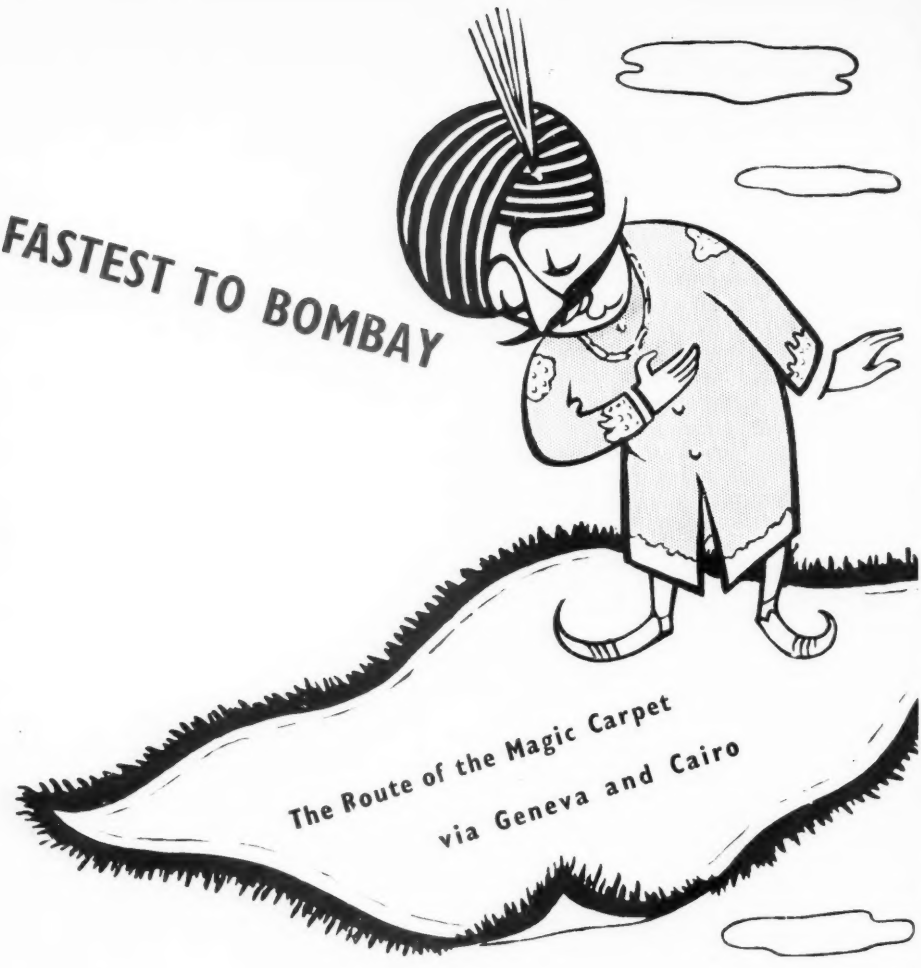
FASTEST TO BOMBAY



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4 HAMILTON PLACE, PICCADILLY, LONDON, W.1

NOVEMBER NOTICES 1948

JOURNAL PREMIUM AWARDS

The Council have set aside an annual sum of £250 for the award of premiums for papers published in the *Journal* and the Council hope that members (or non-members) will contribute papers on their own special subjects.

CONTENTS OF THE NOVEMBER JOURNAL

Some Problems of the Stability and Control of Large Aircraft, by D. J. Lyons, B.Sc.(Eng.), A.F.R.Ae.S.
Rotary Derivatives of a Delta Wing at Supersonic Speeds, by A. Robinson, M.Sc., A.F.R.Ae.S.
Aircraft Servicing, by J. Norman, A.F.R.Ae.S., and S. F. Wilkinson, A.R.Ae.S., M.S.L.A.E.
Why Shear Webs? by H. L. Cox, M.A., F.R.Ae.S.
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ANGLO-AMERICAN AERONAUTICAL CONFERENCE 1949

Arrangements are now being completed for the Aeronautical Conference to be held in 1949.

The Society has been in close consultation with the Institute of the Aeronautical Sciences and a number of lectures have already been arranged. The Conference will spread over five days and will be followed by a number of visits to aircraft firms and to Langley Field and elsewhere.

Recently Mr. A. D. Emil, representing the Institute of the Aeronautical Sciences, paid a visit to England and at a dinner given to him by the Council he explained what steps the Institute were taking to make the Conference an outstanding event in the year 1949 in the history of the Institute. Mr. Emil handed to the Council, on behalf of the Council of the Institute, the following resolution:—

"WHEREAS, it is deemed highly desirable to augment the cordial relations that now exist between the Institute of the Aeronautical Sciences and the Royal Aeronautical Society of Great Britain, and

WHEREAS, the advancement of the aeronautical sciences in the respective countries can best be served by a periodic inter-change of views on various technical matters by representatives of the two Societies,

BE IT THEREFORE RESOLVED, that the President of the Institute of the Aeronautical Sciences, on behalf of the Council of the Institute, extend to the President and Council of the Royal Aeronautical Society an invitation to designate representatives to meet with the Institute of the Aeronautical Sciences in New York City between 22nd May and 6th June or any other dates that may be mutually agreed upon, to discuss problems of common interest in the field of the aeronautical sciences."

The following reply from the Council has been sent to the Institute:—

"The President and Council of the Royal Aeronautical Society welcome the invitation of the President of the Institute of the Aeronautical Sciences to designate representatives to meet with the Institute of the Aeronautical Sciences in New York City between 22nd May and 6th June 1949 to discuss problems of common interest in the field of the aeronautical sciences.

The President and Council are convinced that by such an inter-change of views on technical matters by representatives of the two Societies, the cordial relations which now exist between the Institute of the Aeronautical Sciences and the Royal Aeronautical Society will be enhanced and their objects, namely, the increase in knowledge of aeronautical science and engineering, will be furthered. They have therefore great pleasure in accepting the invitation of the Institute."

The second Conference of the two Societies will have read before it some twenty papers covering a wide range of technical activity, and it is hoped that a good British representation will be possible in addition to those who have accepted the Council's invitation to lecture. The Council feel that the opportunity given by these Conferences to make the personal acquaintance of " opposite numbers " on each side of the Atlantic is one which should be taken every advantage of, as such personal contacts will prove of increasing benefit to aeronautical engineers both in this country and the U.S.A.

Members who propose to attend the Conference should inform the Secretary as quickly as possible.

ASSOCIATE FELLOWSHIP EXAMINATIONS—DECEMBER 1948

The next Associate Fellowship Examinations will be held on 15th, 16th, 17th and 18th December 1948 at the Imperial College of Science, South Kensington, S.W.7. All candidates have been sent a time-table and should make a special note of the times set out for those subjects for which they have entered.

LECTURE PROGRAMME—AUTUMN SESSION 1948

Unless otherwise stated, lectures will be held at 6 p.m. in the Lecture Hall of the Institution of Civil Engineers, Great George Street, London, S.W.1 (by permission of the Council of the Institution). Tea will be served at 5.30 p.m.

Thursday, 11th November 1948

Two French Aircraft Gas Turbines, by M. Destival.

Saturday, 20th November 1948

FULL DAY DISCUSSION ON HELICOPTERS
(A Joint Meeting with the Helicopter Association)

Morning Session

11 a.m. to 1 p.m.—GENERAL PROBLEMS OF THE HELICOPTER FOR CIVIL USE.
11 a.m.—The Operational Point of View, by W/Cdr. R. A. C. Brie, A.F.R.Ae.S., A.F.I.Ae.S.
11.30 a.m.—The Technical Point of View, by Capt. R. N. Liptrot, C.B.E.
12 noon.—Discussion.
1 p.m. to 2.30 p.m.—Luncheon Interval (Members should make their own arrangements).

Afternoon Session

2.30 p.m. to 4.30 p.m.—THE CONSTRUCTORS' APPROACH TO THE PROBLEMS
2.30 p.m.—The Fairey Gyrodyne, by J. A. J. Bennett, D.Sc., M.A., D.I.C., F.R.Ae.S., F.Inst.P.
3.0 p.m.—The Bristol 171 Helicopter, by R. Hafner.
3.30 p.m.—The Cierva Air Horse, by J. S. Shapiro, Dipl.Ing., A.F.R.Ae.S.
4.0 p.m.—Discussion.
4.30 p.m. to 5 p.m.—Tea Interval. (Tea will be provided at the Lecture Hall.)

Evening Session

5 p.m. to 6.30 p.m.—General Discussion and summing up by the Lecturers and the Chairman.

Thursday, 25th November 1948

Development of the Armstrong Siddeley Mamba Engine, by W. H. Lindsey, M.A., A.F.R.Ae.S.

Thursday, 2nd December 1948
Problems in the Development of a New Aeroplane, by G. R. Edwards, M.B.E., B.Sc., F.R.Ae.S.

Thursday, 16th December 1948
Present Thoughts on the Use of Power Flying Controls in Aircraft, by D. J. Lyons.

GRADUATES' AND STUDENTS' SECTION—LECTURES

Wednesday, 10th November 1948—Flying Experiences, by Lt. Cdr. (A) E. M. Brown, R.N., O.B.E., D.S.C., A.F.C., M.A., A.R.Ae.S., Chief Naval Test Pilot.

Wednesday, 24th November 1948—The Investigation of Aircraft Accidents, by Air Cdre. Vernon S. Brown, C.B., O.B.E., M.A., F.R.Ae.S., Chief Inspector of Accidents, Ministry of Civil Aviation.

Tuesday, 7th December 1948—Papers read by members of the Graduates' and Students' Section.

Tuesday, 21st December 1948—Production, The Dynamics of People at Work, by J. V. Connolly, B.E., A.F.R.Ae.S., Professor of Aircraft Economics and Production at the College of Aeronautics.

Meetings will be held in the Library of the Royal Aeronautical Society, 4 Hamilton Place, W.1, at 7.30 p.m. *The Library will not be open before 7 p.m.*

BRANCH NOTICES

BIRMINGHAM BRANCH

Monday, 8th November 1948—Aircraft Armament by S. J. Millars, Boulton Paul Ltd.

At the Birmingham Chamber of Commerce at 7.30 p.m.

COVENTRY BRANCH

Thursday, 18th November 1948—Electronics, by Dr. Whiteley, British Thomson-Houston Ltd.

Thursday, 9th December 1948—Lecture by E. Smith, Research Department, Firth-Vickers Ltd.

All lectures will be held in the Old Gulson Library, Coventry, at 7.30 p.m.

GLASGOW BRANCH

Tuesday, 30th November 1948—" Lecturettes," at 7.30 p.m., at Prestwick Airport.

Thursday, 23rd December 1948—Annual General Meeting, at The Grand Hotel, Charing Cross, Glasgow.

GLOUCESTER AND CHELTENHAM BRANCH

Thursday, 18th November 1948—Problems Facing Civil Air Operations, by N. E. Rowe, C.B.E., F.R.Ae.S.

At the Wheatstone Hall, City Library, Brunswick Road, Gloucester, at 7.30 p.m.

HATFIELD BRANCH

Wednesday, 17th November 1948—Design Problems of Large Flying Boats, by H. Knowler, F.R.Ae.S.

At 6.15 p.m. in the de Havilland Senior Staff Mess.

ISLE OF WIGHT BRANCH

Thursday, 11th November 1948—" Lecturettes "—Four short talks by members of the production staff of Saunders-Roe, Ltd.

Thursday, 2nd December 1948—The Development of the Armstrong Siddeley " Mamba " Engine, by W. H. Lindsey, M.A., A.F.R.Ae.S.

Thursday, 16th December 1948—Annual General Meeting and Film Show.

Meetings will be held in the Sports and Social Club, Saunders-Roe, Ltd., East Cowes, Isle of Wight, at 6 p.m.

LUTON BRANCH

Wednesday, 3rd November 1948—Materials of Aircraft Construction and Material Processing, by Dr. Sutton.

Wednesday, 1st December 1948—The Light Aeroplane and the Future of Private Flying, by P. G. Masefield, M.A., F.R.Ae.S.

Meetings will be held generally at The George Hotel, Luton, at 7 p.m.

PRESTON BRANCH

Wednesday, 17th November 1948—On the Investigation of Aircraft Accidents, by Air Commodore Vernon Brown, C.B., O.B.E., F.R.Ae.S.

Wednesday, 8th December 1948—Gusts, by Professor A. A. Hall, M.Sc., M.A., F.R.Ae.S

All Lectures will be held in the Assembly Hall of the Technical College, Corporation Street, Preston, at 7.15 p.m.

PORTSMOUTH BRANCH

Friday, 26th November 1948—Model Section Display.

Friday, 3rd December 1948—Films showing " How an Aeroplane Flies."

Friday, 17th December 1948—Some Aspects of Carrier Operation, by Capt (E) B. H. Cronk, D.S.C., R.N.(Retd.).

All lectures will be held in the Lecture Hall at the Central Library, Guildhall Square, Portsmouth, at 7 p.m.

SOUTHAMPTON BRANCH

Wednesday, 10th November 1948—Aircraft Production, by S. E. H. Lefever.

Wednesday, 8th December 1948—Rocket Propulsion and Interplanetary Flight, by A. V. Cleaver, A.R.Ae.S.

WEYBRIDGE BRANCH

Wednesday, 17th November 1948—Some Aspects of Practical Flying, by J. K. Quill, O.B.E., Vickers-Armstrongs Ltd. (Supermarine).

Wednesday, 1st December 1948—The Island Campaign (Coloured film), shown by W. Courtenay, M.M., A.R.Ae.S., *Daily Graphic* Aeronautical Correspondent.

Wednesday, 15th December 1948—Annual General Meeting, followed by films of Vickers Aircraft.

Meetings will be held at Vickers-Armstrongs, Ltd., Weybridge Works, at 6 p.m., unless previous notice is given.

AERODYNAMICISTS WANTED

Aerodynamicists are wanted by the Ministry of Supply for posts in London in Senior Scientific Officer and Scientific Officer grades.

Candidates must have 1st or 2nd Class Honours degree in mathematics, physics or engineering. Experience in one of the following fields is desirable (*a*) administration of aeroynamics research work, (*b*) estimation of aircraft performance, involving knowledge of I.C.A.O. regulations and their significance in aircraft design, (*c*) formulation and revision of I.C.A.O. performance requirements, (*d*) civil aircraft projects including route analysis and operating costs.

Inclusive salary ranges are:—

Senior Scientific Officer ..	£700-£900
Scientific Officer	£400-£650

Rates for women are somewhat lower.

The appointments are unestablished but carry F.S.S.U. benefits.

Write quoting A.364/48A to Ministry of Labour and National Service, Technical and Scientific Register (K), York House, Kingsway, W.C.2, for application forms which must be returned completed by 30th November 1948.

LAPEL BADGES

Small lapel badges of the Society's crest in metal silver-plated, and in light and dark blue enamel, will be available shortly for members. The cost of each badge will be 3/6 (including postage).

CHANGES OF ADDRESS

To assist in keeping the records of members correct and up to date the Secretary will be glad if all members will notify him as soon as possible of changes of address.

When notifying changes please give the following particulars:—

Name (in block letters).
Grade of membership.
New address (in block letters).
Old address.

Changes of address must be received *before the 15th of the month* in order to be effective for the Journal for the following month.

NEW MEMBERS

Associate Fellows

Frank Bernard Baker, Herbert Charles Barton, Joseph William Birkle, Arthur Joseph Kenneth Carline, Edgar Theodore Carpenter, Walter Leonard Clarke, Ernest Charles Covell (from Associate), Horace Richard Davenport, William Henry Dresser, Neville John Ebsworth (from Associate), Reginald Joseph Fenner, Sydney Harrison-Smith, Frederic Charles Haus, William Heaton, Herbert William Heslop, George Kenyon, Leslie Robert Knight (from Graduate), Ronald Stephen Loveland, Dennis John Lyons, James Fraser Montgomerie, Constantine Morley, Jethro Jack Canham Oakley, Leslie Thomas Palmer (from Associate), Donald Westwood Richardson (from Associate), William Sutcliffe, John Bernard Joseph Thorpe.

Associates

James Robert Anderson, Arthur George Herbert Angell, Charles Lewis Baker, Peter Bennison (from Student), Victor Francis Bingham, Frederic Lister Chadwick, Walter Patrick Copinger, Kenneth Costall, Roland Arthur Victor Dismore, Bertram James Elkins, Arnold Field, John Frederick Gardner, John Kenneth Hadwin, Bernard John Hammett, Hawkesley Murray Arthur Hayes, Robert Leslie Hughes, Basil Barry Jupp, Robert King, Norman Frederick Lamb, Sidney William George Marchant, Edgar William Lovejoy, Leslie Milnes, Douglas Robert Morgan, Harold Lewis Newman, Frederick Thomas Pankhurst, Albert Henry Ridley, Raymond Frederick Sargent (from Student), Edward Elijah Weekley, Gilbert Frank Wilkinson, Harold Wilson.

Graduates

Denis William Bryer, Ronald Frederick Cash, Norman Stanley Currey, Pierre Robert Genot, Kenneth Herbert Griffin (from Student), Harold Frederick Hawkins, Thomas McCloghry, Stanislaw S. Schaetzel (from Student), George Tolmie Skinner (from Student), Arthur Montgomerie Thompson (from Student), Richard Phillip Wade.

Students

John Cabread Hall, Peter Milns, Charles Roy Pratt-Barlow, Robert Charles Leslie Sample, George Thompson, John Thompson

Companions

Jamsked Minocher Cowasjee (from Student), William Thomas Fogarty, Kenneth Stuart-Smith.

ACKNOWLEDGMENTS

The Council acknowledge with grateful thanks the gift of books from the Royal Aero Club and from H. J. Wolffsohn, Esq.

ADDITIONS TO THE LIBRARY

Pamphlets in italics with location reference following in brackets. Books marked * or ** may not be taken out on loan.

BB.a.16—Electric Resistance Strain Gauges. W. B. Dobie and P. C. G. Isaac. English Universities Press. 1948.

D.b.278—World Aviation Annual 1948. J. Parker van Zandt. (Ed.) Aviation Research Institute. 1948.

E.b.131—Fundamentals of Vibration Studies. R. G. Manley. Chapman & Hall. 1948.

E.b.132—Engineering Mechanics. S. Fairman and C. S. Cutshall. John Wiley & Sons. 1948.

E.g.11—Centrifugal and Axial Flow Pumps. A. J. Stepanoff. John Wiley & Sons. 1948.

EE.h.37—Gas Turbines and Their Problems. Hayne Constant. Todd Publishing Co. 1948.

G.a.83—Essential Metallurgy for Engineers. A. C. Vivian. Pitmans. 1948.

G.e.F.34—Aircraft Welding. R. H. Drake. Macmillan, New York. 1947.

G.e.J.32—Non-ferrous Metals and Alloys. E. Gregory and E. N. Simmons. Paul Elek. 1948.

G.e.J.33—Aircraft Sheet Metal. R. H. Drake. Macmillan, New York. 1947.

I.c.37—Pure Mathematics, Part I. Wilhelm Süss. F.I.A.T. 1948.

J.g.217—Weather Forecasting. S. W. C. Pack. Longmans. 1948.

L.d.123—*Notices to Airmen. Nos.* 326-329, 331-345, 347, 349, 350, 352-354, 356-364.

N.a.92—One, two, three—infinity. G. Gamow. Viking Press, New York. 1947.

*X.a.59—The Wright Aero Handbook. Wright Aeronautical Corporation. 1946.

A.R.C. Reports and Memoranda

2080—*The estimation of wing structure weight. W. Tye and P. E. Montagnon.*

2118—*Theory of the flat elastic parachute. W. J. Duncan, G. W. Stevens and G. J. Richards.*

2221—*Dynamic loads in aeroplanes under given impulsive loads with particular reference to landing and gust loads on a large flying boat. D. Williams and P. N. Jones.*

2236—*Air resistance of a flat plate of very porous material. G. I. Taylor.*

2244—*The boundary-layer flow over a permeable surface through which suction is applied. J. H. Preston.*

2260—*Comparative effects of small changes in profile drag and weight on aircraft performance. A. L. Courtney.*

2261—*The lift increment of an aerofoil due to variation of incidence along the span, and a simple method of estimating the lift distribution. M. Gdaliahu.*

2268—*Flight tests on the longitudinal behaviour of the Vengeance IV with special reference to the characteristics of the spring tab elevator. J. Nivison.*

2280—*The design of minimum drag inboard fins. V. M. Falkner.*
With appendix: On the conformal transformation of a wing with symmetrical inboard fins. Sir Charles Darwin.

2282—*Tearing experiments on metal sheets. P. W. Rowe and C. Gurney.*

2283—*The effect of radial pressure on the flow and fracture of reinforced plastic rods. C. Gurney and P. W. Rowe.*

2285—*The effective stress concentration at the end of a crack having regard to the atomic constitution of materials. C. Gurney.*

2288—*The estimation of wing divergence speeds. E. G. Broadbent.*

2290—*The rolling vibration of an aircraft. J. Morris and D. Morrison.*

2291—*The pitching vibrations of an aircraft. J. Morris and G. S. Green.*

2296—*Measurements of cabin noise in bomber aircraft. D. Cameron and W. J. D. Annand.*

2301—*Tests on four circular-back aerofoils in the compressed air tunnel. D. H. Williams, A. F. Brown and C. J. W. Miles.*

N.A.C.A. Technical Reports

756—*The induction of water to the inlet air as a means of internal cooling in aircraft-engine cylinders. A. M. Rothrock, A. Krsek, Jr., and A. W. Jones.*

758—*Performance of NACA eight-stage axial-flow compressor designed on the basis of airfoil theory. J. T. Sinnette, Jr., O. W. Schey and J. Austin King.*

N.A.C.A. Technical Memoranda

1192—*Rotating disks in the region of permanent deformation. F. Laszlo.*

N.A.C.A. Technical Notes

1450—*An investigation of aircraft heaters. XXVI—Development of a sensitive plated-type thermopile for measuring radiation. L. M. K. Boelter, E. R. Dempster, R. Bromberg and J. T. Gier.*

1451—*An investigation of aircraft heaters. XXVII—Distribution of heat-transfer rate in the entrance section of a circular tube. L. M. K. Boelter, G. Young and H. W. Iverson.*

1550—*Tensile and compressive properties of laminated plastics at high and low temperatures. J. J. Lamb, I. Boswell and D. M. Axilrod.*

1551—*Fracture strength of 25S-T aluminium alloy under combined stress. E. G. Thomsen, I. Lotze and J. E. Dorn.*

1580—*X-ray diffraction investigation of minor phases of 20 high-temperature alloys. B. M. Rosenbaum.*

1592—*Compressible flow tables for air. M. A. Burcher.*

1610—*Effect of screens in wide-angle diffusers. G. B. Schubauer and W. G. Spangenburg.*

1619—*An evaluation of proposed reference fuel scales for knock rating. H. C. Barnett and T. C. Clarke.*

1632—*Gust-tunnel tests to determine influence of airfoil section characteristics on gust load factors. B. H. Pierce and Mitchell Trauring.*

1654—*Sound from dual-rotating and multiple single-rotating propellers. H. H. Hubbard.*

1655—*Thermodynamic charts for the computation of fuel quantity required for constant-pressure combustion with diluents. D. Bogart, D. Okrent and L. Richard Turner.*

1656—*Tests of six types of bakelite bonded wire strain gages. W. R. Campbell.*

1657—*Effects of compressibility on the flow past thick airfoil sections. B. N. Daley and M. D. Humphreys.*

1658—*Lateral stability and control characteristics of a free-flying model having an unswept wing with an aspect ratio of 2. M. O. McKinney, Jr., and R. E. Shanks.*

1659—*Method for calculation of pressure distributions on thin conical bodies of arbitrary cross section in supersonic stream. S. H. Maslen.*

1660—*Characteristics of thin triangular wings with constant-chord partial-span control surfaces at supersonic speeds. W. A. Tucker and R. L. Nelson.*

1661—*Critical axial-compressive stress of a curved rectangular panel with a central chordwise stiffener. S. B. Bardorf and Murry Schildcrout.*

1662—*Aerodynamic properties of slender wing-body combinations at subsonic, transonic and supersonic speeds. J. R. Spreiter.*

1663—*High-speed wind-tunnel investigation of an NACA 65-210 semi-span wing equipped with plug and retractable ailerons and a full-span slotted flap. J. Fischel and L. R. Schneiter.*

1666—*Flight investigation of effects of rotor-blade twist on helicopter performance in the high-speed and vertical-autorotative-descent conditions. A. Gessow.*

1667—*Effect of strength and ductility on burst characteristics of rotating disks. A. G. Holms and J. E. Jenkins.*

1672—*Supersonic wave drag of non-lifting sweptback tapered wings with mach lines behind the line of maximum thickness. K. Margolis.*

1671—*Effect of taper ratio on low-speed static and yawing stability derivatives of 45 degrees sweptback wings with aspect ratio of 2.61. W. Letko and J. W. Cowan.*

1674—*Estimation of effectiveness of flap-type controls on sweptback wings. J. G. Lowry and E. L. Schneiter.*

1675—*Temperature gradients in the wing of a high-speed airplane during dives from high altitudes. T. Tendeland and B. A. Schlaff.*

1678—*Graphical method of obtaining theoretical lift distribution on thin wings at supersonic speeds. C. B. Cohen and J. C. Evvard.*

1682—*Heat transfer to bodies travelling at high speed in the upper atmosphere. J. R. Stalder and D. Jukoff.*

1683—*An experimental investigation of an NACA 6311012 airfoil section with leading-edge suction slots. G. B. McCullough and D. E. Gault.*

The following reports have also been received:—

Nationaal Luchtvaartlaboratorium, Amsterdam

Reports F.28, F.32 and *S.340.*

C.S.I.R. Division of Aeronautics

Report SM.107.

Publications Scientifiques et Techniques du Ministere de l'Air

Report No. 218.

J. LAURENCE PRITCHARD,
Secretary.

THE LEWES PRESS (Wightman & Co. Ltd.), Friars Walk, Lewes, Sussex.

The Royal Aeronautical Society

ACTIVITIES OF THE GRADUATES' AND STUDENTS' SECTION

November 1948.

VISITS

On Tuesday, 20th July 1948, a party of members of the Section visited the Royal Aircraft Establishment, South Farnborough, Hants. The programme included a display of films illustrating the work of the Establishment, a tour of the airfield and hangars and visits to the X-ray section of the Metallurgy department, the airframe test-rigs (the "Cathedral" and "Temple") of the structures department, the low-speed, high-speed and supersonic wind tunnels, and the wind tunnel model-makers', carpenters' and fabric shops.

On Saturday morning, 14th August 1948, a visit was made to the Aerodynamics Division of the National Physical Laboratory, Teddington, Middlesex. Tours were made of the low-speed, high-speed, supersonic and compressed-air wind tunnels, and of the whirling arm.

The works of Vickers-Armstrongs Ltd., Weybridge, Surrey, were visited on Saturday morning, 28th August. This visit consisted of a tour of the factory, during which members were shown the assembly line for the production of Valetta aircraft, the machine shop, and the tinsmiths', welders' and wing shops.

A visit to the Ford Motor Co., Dagenham, Essex, was made on Wednesday, 8th September 1948, at 2 p.m. The party saw raw materials (coal and iron ore) being unloaded at the Ford wharves, and followed these to the coke ovens, blast furnaces, melting plant and electricity generating plant. The sand mould shop, rolling mills and machine shop were then visited. The engine production belt was next seen, the assembly of V8, four-cylinder and tractor engines being viewed from a bridge over the shop. The tractor conveyor was followed down through the engine assembly stage to the final assembly immediately before painting. The V8 and ten-hundredweight vans were followed along the production line where the prefabricated chassis received bodywork (from Messrs. Briggs of Dagenham), axles, wheels, engine, and wiring. On the adjoining belt were the Anglia, Prefect and American Anglia assemblies. The guide informed the party that the Ford works produced 550 vehicles a day. The visit ended with tea, provided by the Ford Company.

The last visit of the year was made on Saturday, 25th September 1948, to Northolt Airport, Ruislip, Middlesex. After assembling at the entrance, members were taken by coach to the control tower, where the control system was explained in some detail. A tour was made of the B.E.A.C. hangars and workshops, where the servicing docks for Viking aircraft were inspected and the instrument and starter motor test benches. After lunch, the meteorological and briefing sections were visited, and the functions and methods of operation of the sections were explained and demonstrated. As practice GCA approaches were being made, a visit arranged to the Ground-Controlled Approach unit had to be cancelled, and the plotting unit, which could keep track of all aircraft within a thirty-mile radius, was seen instead. The passenger buildings, customs rooms, waiting rooms and passengers' restaurant were then seen, and after being mistaken for passengers from Jersey, the party was taken to the public enclosure, where the visit ended at 5.30 p.m.

The grateful thanks of the Section are due to the Directors of the establishments visited during the year, for permission to do so, and also to those able and willing persons who arranged the programmes, acted as guides and answered questions with such patience and good humour.

LECTURE

The first lecture of the Autumn session was held at the offices of the Society on Tuesday, 28th September 1948, at 7.30 p.m. The Section Chairman, Mr. J. W. F. Housego, presided, and introduced the lecturer, K. G. Wilkinson, Esq., B.Sc.,

A.C.G.I., D.I.C., A.F.R.Ae.S., of British European Airways Corporation, who read a paper on "Sailplane Design."

Mr. Wilkinson began by explaining that the design of a sailplane was a matter of arriving at the best compromise with the forces of nature and involved no man-made requirements. Design had advanced with the designer's understanding of the natural requirements. Initially, the only need was for a low sinking speed, since the early stages of gliding involved jumping from high places and drifting down. This led to light structures and highly cambered wings, but did not result in particularly good control characteristics. The second stage, slope soaring, called for better control and the consequent efforts to cross gaps in the soaring slopes led naturally to a demand for a good gliding angle. The technique of thermal flying which was discovered first during slope soaring flights, required a low sinking speed, good control, and the ability to circle safely near the stall. From this stage, cross-country flying had evolved, which required sailplanes capable of high cruising speeds. At the present time there was a need for specialised designs, with pressure cabins, for high altitude wave flying, and a specialised mountain-soaring type with moderate span and high manœuvrability to take advantage of the narrow bands of lift in mountainous regions.

The development of the sailplane had been empirical and the evolution of the most suitable type had been the result of competition flying. The most generally useful type was the thermal flying sailplane, which could cope with hill soaring with an efficiency almost equal to that of the earlier types, and had the added advantages of higher manœuvrability and better glide ratio.

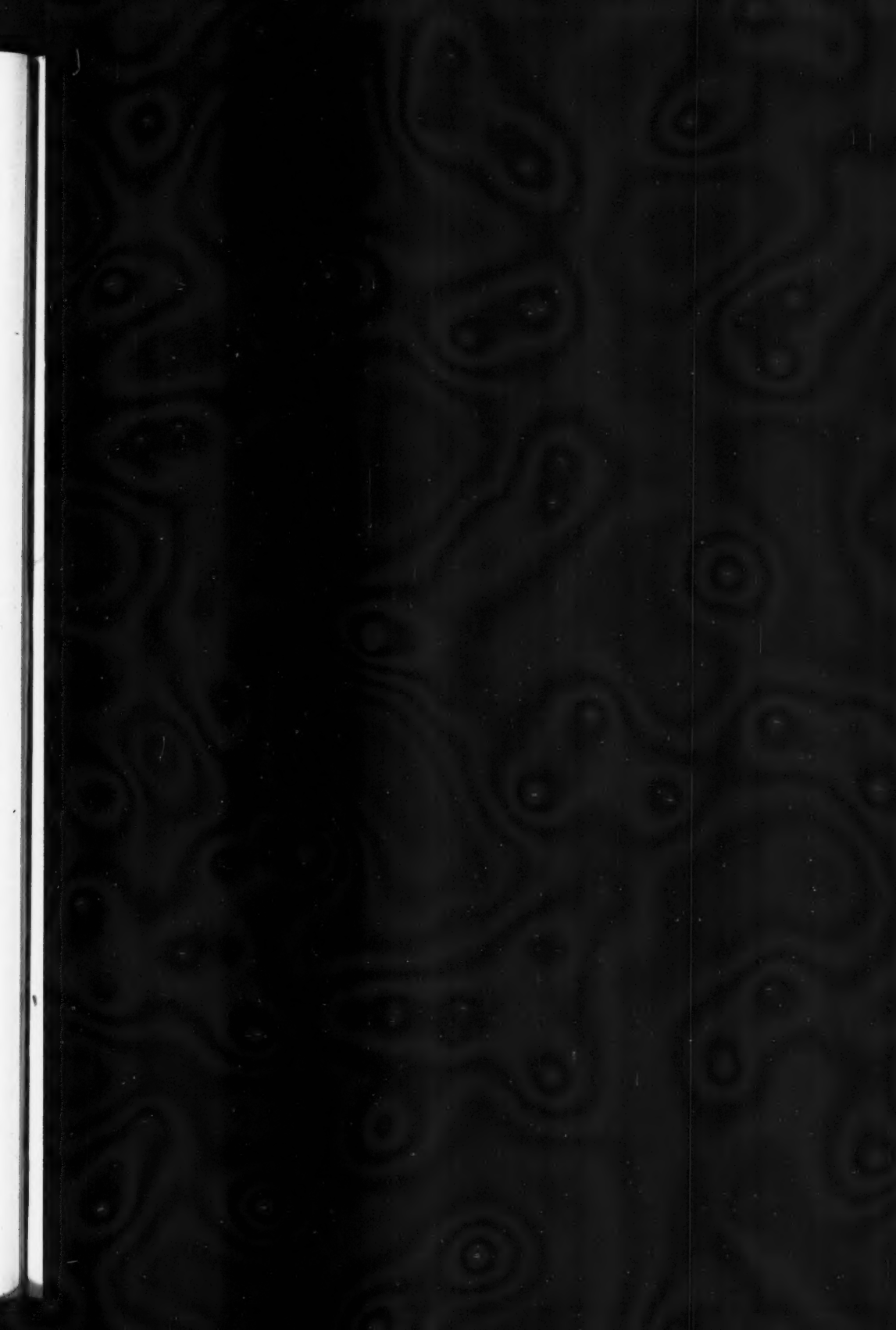
The lecturer proposed to begin with the basic elements of design and show how the answer was obtained. A number of slides were shown which summarised performance and weight data for a number of types which had been tested in this country and in Germany, and demonstrated the consequences of varying the main design parameters of span and aspect ratio by analysis of this data. Relative performances in various thermal conditions were considered, and from this, a number of conclusions could be reached. Depending on the thermal strength, different design parameters had the predominant effect. With low strength thermals, high span and relatively moderate aspect ratios were the most efficient; with high strength thermals, the effect of span became less noticeable, but the effect of aspect ratio was more pronounced, and should be not less than 20. It was noticed that flaps had little effect except on sailplanes with short span and high aspect ratio, and even then had to be exceedingly well designed.

The development of a moderate area of laminar flow was important, and if the transition point was moved from 20 to 40 per cent. of the chord, this had more effect than changing the span from 60 to 70 feet, or the aspect ratio from 25 to 30, both of which tended to increase the cost of the sailplane. The main compromise was between expense and performance. The best performance was required for a given price, which meant that the weight, and hence the span, had to be kept low, and that extremely high aspect ratios had to be avoided if these involved exotic structural methods. The effect of weight change on a given sailplane, by the use of ballast, had little effect unless the weight was increased by a large amount, which meant that it had to be distributed along the wing.

Mr. Wilkinson also explained the principles of inter-thermal flying, showing how the speed at which to fly in order to lose least height could be found. A number of colour slides of cloud formations and of existing sailplanes, and a number of black-and-white slides of various sailplanes were shown, and the characteristics of each type outlined.

NOTICE

With the resignation in August of Mr. G. B. Griffiths, a vacancy arose on the Section Committee. The Committee has exercised its right to co-opt another member of the Section and has asked Mr. A. Stanbrook, of the Fairey Aviation Company to fill this post. Mr. Stanbrook, who is a Student of the Society, was one of the unsuccessful candidates in the election of officials at the Annual General Meeting in April last.



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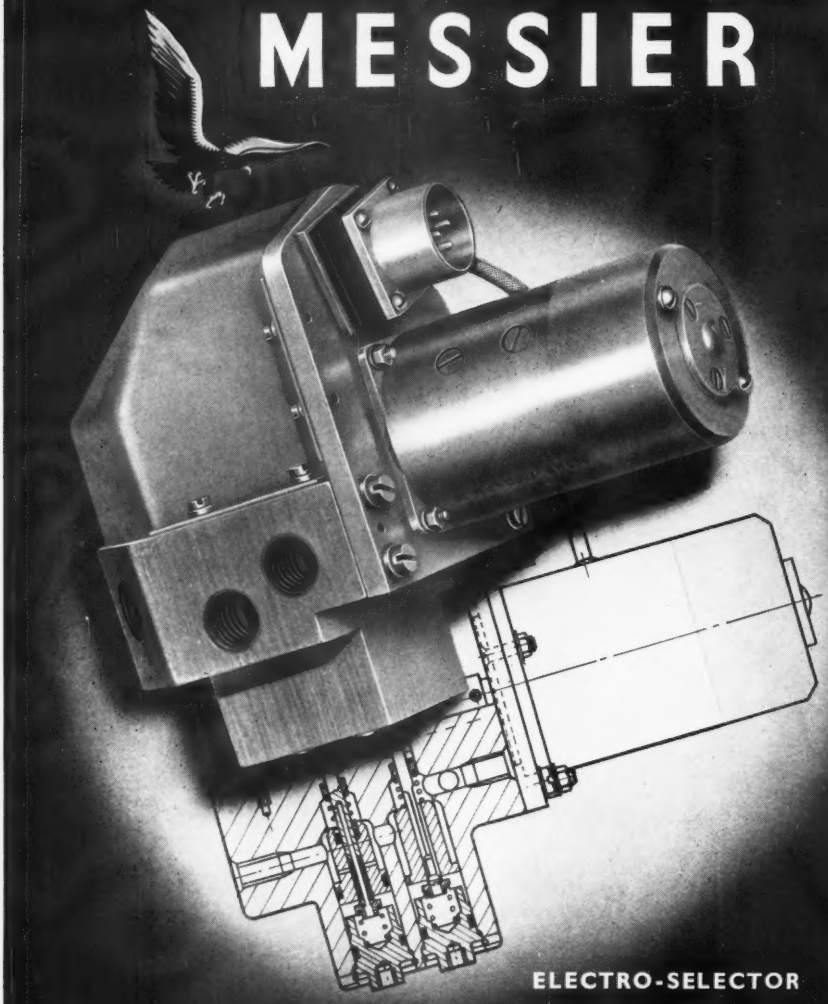

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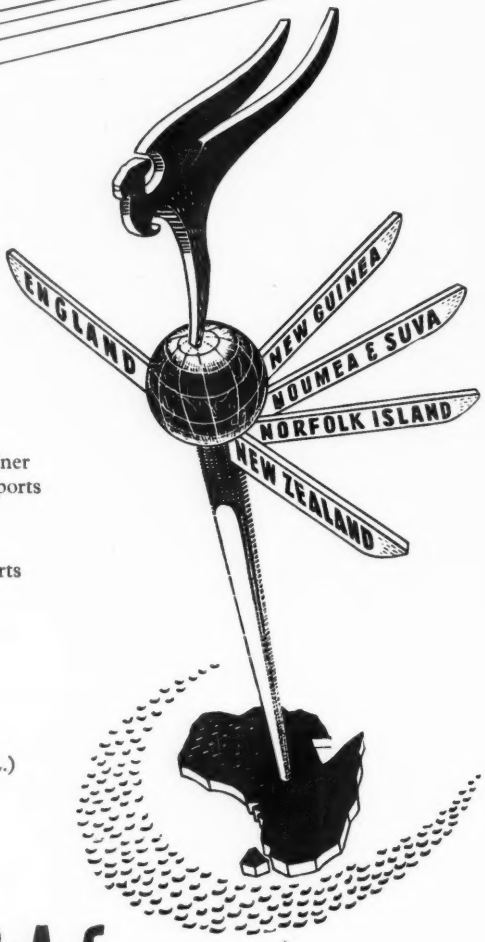

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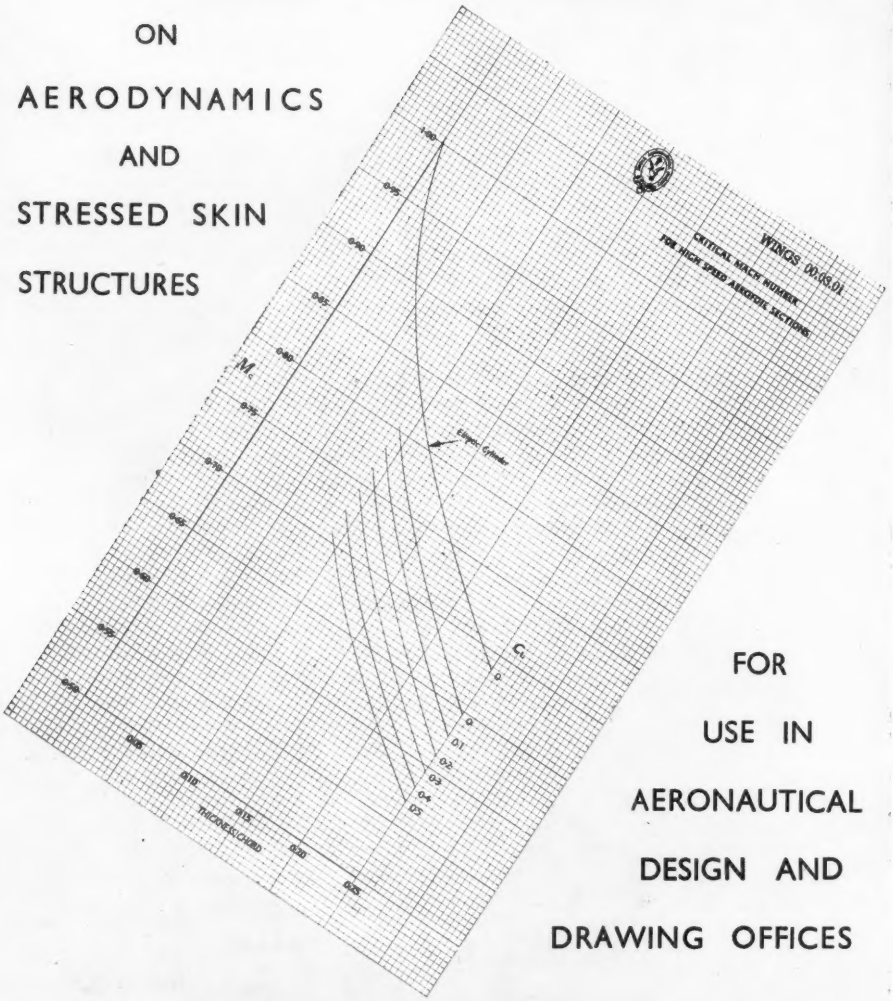

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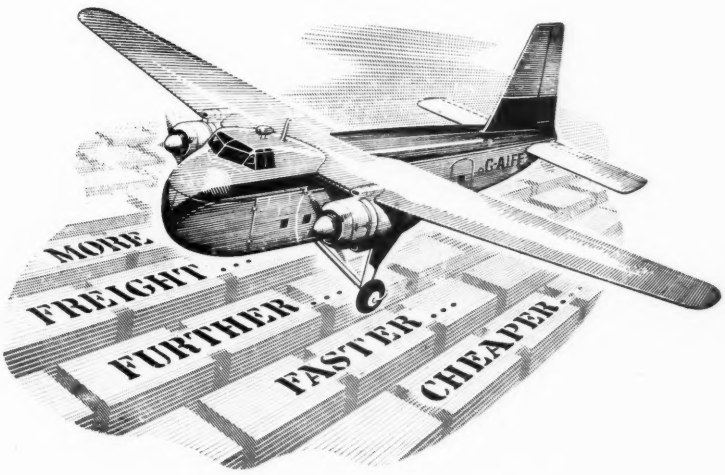

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Awarded for the best paper read in any year before the Society on any science allied to aeronautics, *e.g.*, meteorology, wireless telegraphy, instruments.

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Awarded under the same conditions as those for the Silver Medal, but for some less important advance in aeronautical design.

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The Wilbur Wright Memorial Lecture is held annually, a premium of £75 being awarded to the lecturer invited by the Council to deliver the lecture. The lecture is usually given alternately by an American and and Englishman, and is the most important aeronautical lecture of the year.

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Only very simple symbols and formulæ should be typewritten. All others should be written carefully by hand in ink. Ample space for marking should be allowed above and below all equations. Greek letters should be designated by name in the margin.

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SOME PROBLEMS OF THE STABILITY AND CONTROL OF LARGE AIRCRAFT

by

D. J. LYONS, B.Sc. (Eng.), A.F.R.Ae.S.*

IN THE REALM of stability and control, we are at present travelling headlong into a region of new problems. Besides tidying up the outstanding items on the types of aircraft we are used to seeing flying about to-day, a not inconsiderable task when one considers the troubles we encounter when we undertake the design of a new so-called "conventional" aeroplane, we have to tackle two major lines of research and development. First, the peculiar problems associated with the stability and control of aircraft of large size, both military and civil, into which difficulties associated with high Mach numbers intrude themselves but little; and secondly, the even more extensive problems associated with the stability and control of aircraft, both large and small, which are to fly at very high Mach numbers, not only at their design cruising and diving speeds, but also in the extremely important slow speed conditions. Mr. M. B. Morgan of the R.A.E. has dealt with some aspects of the second series of problems,† so I propose to concentrate on some of the problems involved in the first series, that is those primarily associated with the increase of aircraft size.

First it may be as well to state that I include under the general term "large aircraft," all aircraft of, say, 100 ft. wing span and above with a wing loading above say, 30-40 lb. per square foot. This statement may be surprising, as aircraft at the lower limit of this range have been flying for some considerable time, but there is little doubt in the minds of most people who have had to deal with this problem that the solutions of the control problems as applied to the majority of these aircraft are far from satisfactory. Most of these aircraft with ordinary fixed balance controls have controls which are either too heavy or too sensitive to production and other variations. The pilots of heavy aircraft have learnt to accept heavy controls, but this heaviness limits considerably the manœuvring ability and safety during take-off, approach and landing.

1. THE DYING OUT OF FIXED BALANCE.

In Fig. 1 some curves are given which illustrate the position at which we have arrived with direct manually-operated fixed balance controls; the term "fixed balance" includes nose balance of any form, horn balance and geared tabs, in fact any kind of balance which remains unchanged with speed. Against the aircraft's wing span has been plotted the maximum Kb_2 (*i.e.* the net rate of change of control hinge moment coefficient with control angle including response effects) which is required, on each of the three controls of a typical conventional aircraft, in order to give a pilot's force equal to, or less than, an amount considered desirable in four different flight conditions.

Three of the conditions, one for each control, are associated with the approach, landing and take-off, and these are probably the most severe cases for civil aircraft. The elevator and aileron cases are for the approach in bumpy conditions, and the rudder for the engine cut on take-off. The approach and take-off cases are also generally the most severe for the elevator and rudder controls of the heavy military aircraft, but a more severe high speed manœuvring case has been added for the ailerons of this type of aircraft. To illustrate the effect of size without other complication, the aircraft controls

Lecture given to the Graduates' and Students' Section on 10th February 1948.

* Mr. Lyons is Principal Scientific Officer in the Aerodynamics Flight Section at the Royal Aircraft Establishment.

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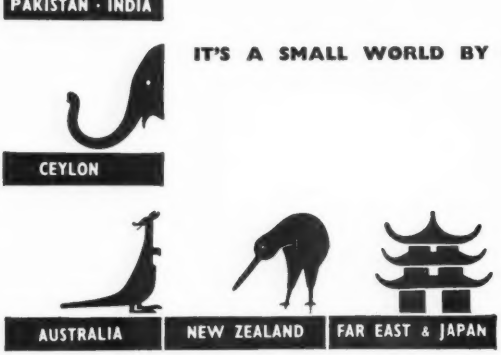

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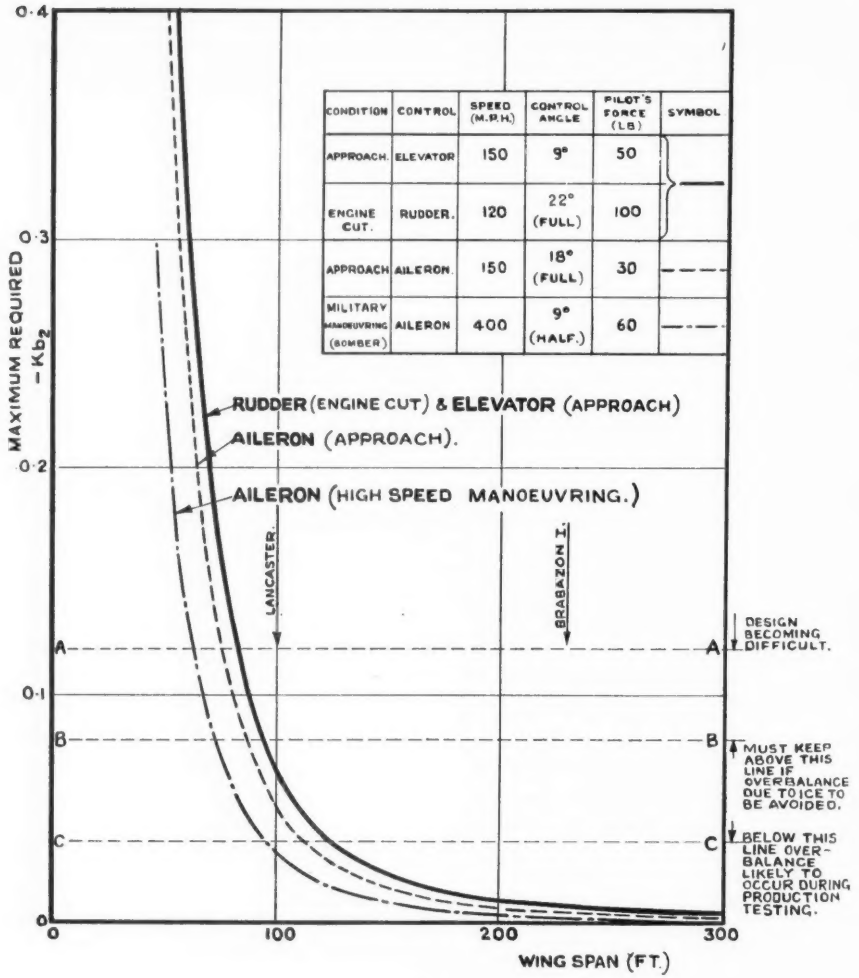
0·4
0·3
0·2
0·1
0
MAXIMUM REQUIRED $-Kb_2$
CONDITION | CONTROL | SPEED (M.P.H.) | CONTROL ANGLE | PILOT'S FORCE (LB) | SYMBOL
APPROACH. | ELEVATOR | 150 | 9° | 50
ENGINE CUT. | RUDDER. | 120 | 22° (FULL) | 100
APPROACH | AILERON. | 150 | 18° (FULL) | 30
MILITARY MANOEUVRING (BOMBER) | AILERON | 400 | 9° (HALF.) | 60
RUDDER (ENGINE CUT) & ELEVATOR (APPROACH)
AILERON (APPROACH).
AILERON (HIGH SPEED MANOEUVRING.)
LANCASTER.
BRABAZON I.
A
B
C
DESIGN BECOMING DIFFICULT.
MUST KEEP ABOVE THIS LINE IF OVERBALANCE DUE TO ICE TO BE AVOIDED.
BELOW THIS LINE OVER-BALANCE LIKELY TO OCCUR DURING PRODUCTION TESTING.
0
100
200
300
WING SPAN (FT.)

Fig. 1.

Typical variation with size of required Kb_2 of control to meet various specified flight conditions with manual control.

are assumed to be scaled up linearly with the wing span; the basic aircraft assumed at a wing span of 100 ft. is something like a Lancaster with rather larger elevator and rudder controls, and to keep the approach speeds constant the wing loading has been assumed constant at about 60 lb./sq. ft.

It is apparent for all four cases, none of which is over-severe, that the required maximum value of Kb_2, falling off as it does inversely as the cube of the linear dimensions, quickly reaches a quite impractical value. It helps, in assessing the difficulty or the impossibility of the problem for any given size of aircraft, to place on the figure three limit lines AA, BB and CC. Below AA, which is drawn on the basis of experience at a Kb_2 of -0.12, the difficulty of design of the control is becoming intense; that is, it is difficult for the design of the control to be settled by existing knowledge, both theoretical and empirical, and it is fairly certain that exten-

sive modifications will have to be made during the prototype stage to achieve the desired value of Kb_2.

Line BB, which is drawn at a Kb_2 of -0.08, represents the minimum value of Kb_2 which is required to ensure that the control is not over-balanced by the effects of ice accretion along the leading edge of the aerofoil surface; -0.05 of this limit is required by the effects of the ice accretion itself, and the rest to cover manufacturing variations in the basic control hinge moments. Line CC, which is drawn at a Kb_2 of -0.035, represents the minimum Kb_2 which can be worked to, without acute danger of over-balance occurring on the controls of production aircraft because of manufacturing tolerances.

The difficulties we have experienced in satisfactorily balancing the controls of an aircraft of the Lancaster size are well illustrated; in fact, on one aircraft of this size over-balance of the ailerons was met during production testing, fortunately only at high speeds, so that the condition could be approached cautiously. Indeed it is clear we are already over the boundary of really practical engineering, and for aircraft of the size of the Brabazon I, the problem of control balance is obviously insoluble with fixed balance and manual operation.

How are we to deal with the problem? There are three main lines of attack put forward:—

(*a*) to use an aerodynamic trick balance, such as a spring tab, which varies its balance with manufacturing errors, or

(*b*) to use a pure servo tab, that is with the pilot's control connected directly with the tab, and the tab supplying the power to drive the main control surface. There might or might not be follow-up between the tab and the control surface: or

(*c*) to use powered flying controls.

2. SPRING TAB CONTROLS.

The basic principle of the spring tab system is that the pilot's effort is directed into the control surface through a spring member, and a tab on the control surface is moved, in a balancing sense, an amount proportional to the deflection of this spring. This system, in effect, gives the pilot at each flight speed, a certain fraction of the control force that he would have had with the spring tabs locked; this fraction changes with flight speed, however, and by suitable choice of the geometry, the aerodynamic balance of the tabs and the stiffness of the circuit from the spring to the tabs, the way it changes with speed can be varied within fairly wide limits. Thus the spring tab is unlike all forms of fixed balance which, in effect, give the pilot the difference between two large forces, both of which are subject to manufacturing variations. This has made the spring tab of considerable use in enabling controls to be manually operated on faster and larger aircraft than was previously possible. Instead of being limited to the values indicated in Fig. 1, we are able to operate with any value of Kb_2 desired, as long as the control surface with spring tabs locked is sufficiently unbalanced to avoid the ice accretion and manufacturing tolerance difficulties. In practice it is found sufficient to work to a Kb_2 with tabs locked of between -0.2 and -0.3, so that we are well clear of these troubles, and moreover in the region where design of the control balancing is relatively simple.

Let us examine, however, the way in which the spring tab control will cope with increasing aircraft size. The control surface is driven by the combined action of the aerodynamic forces caused by the tab movement, the moment of the force in the tab operating arm about the control hinge and the moment of the force in the spring about the control hinge, the forces in the tab rod and spring being supplied by the pilot. In Fig. 2 the percentage of the control hinge moments with spring tabs locked that is supplied by the spring is plotted against the wing span of the "typical" aircraft, for the aileron approach case. Two follow-up ratios between the tab and control have been used, a value of 3:1 which is in common use for spring tab controls and a low value of 1:1.

The term "follow-up ratio," mathematically, is the apparent gearing of the tab to the main control surface, when the main control is moved and the pilot's control held fixed; the ratio is approximately equal to X/Y in the thumbnail sketch in Fig. 2, the tab moving, of course, in an anti-balance sense. It has important repercussions on the aerodynamics and flutter propensities of all tab systems; the higher the follow-up ratio, the greater the gearing has to be between the pilot's control and the tab, and hence the greater the pilot's force necessary to drive the tabs; also, the more difficult it is to meet the anti-flutter criteria.

High follow-up ratios, *i.e.* 3:1 and above, usually have to be used when spring tabs are

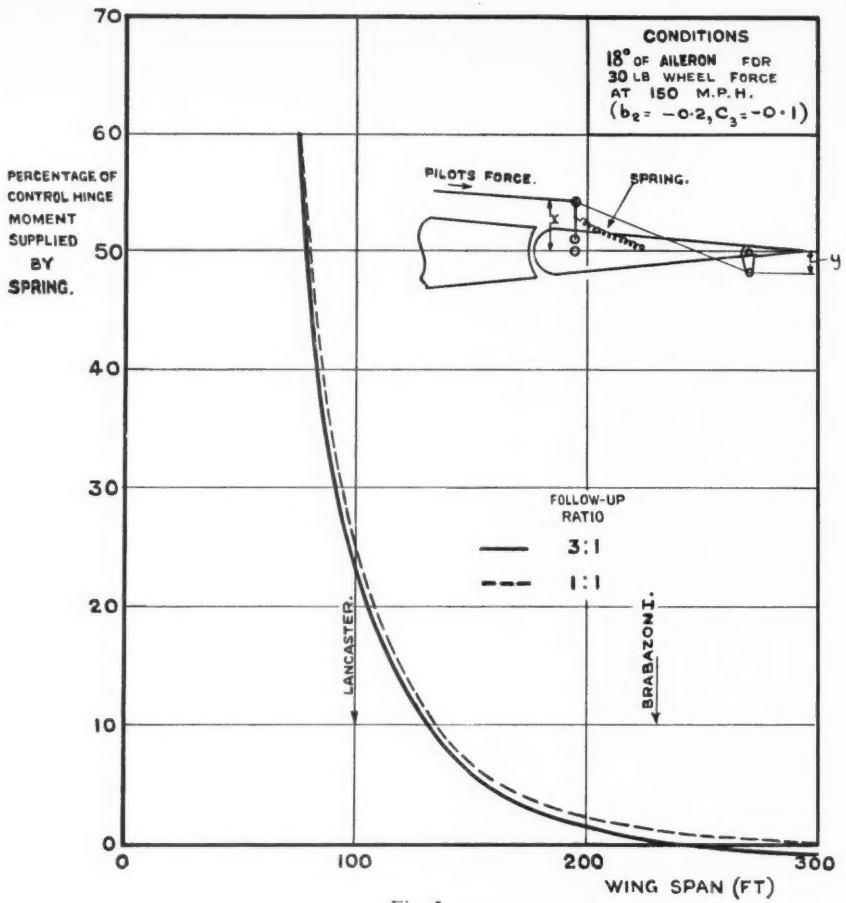

Fig. 2.
Variation with size of the proportion of the control hinge moment supplied by the spring in a typical spring tab control designed to meet a given flight condition.

fitted retrospectively to existing controls, either to avoid large changes in the gearings throughout the control circuits or to avoid large losses in control surface movements. There is also some belief that a feeling of "sponginess" in the control on the ground and at low flight speeds is reduced to a minimum by high follow-up ratios; in conjunction with a relatively unbalanced spring tab, it also enables the elevator control to be made heavier at high speeds as opposed to the usual action of the spring tab in "beating the speed squared law." On the other hand, there does not appear to be any objection to a very low ratio other than an accusation of "sponginess" of control, and as servo tab controls have flown satisfactorily in which the spring stiffness has effectively been reduced to zero and on which there is complete freedom between stick and control surface on the ground, it does not appear that this objection should be serious.

Two points become apparent immediately on examining Fig. 2:—

(*a*) with aircraft spans above 150 ft. or so the spring, even at the relatively low landing speeds, is doing little in helping to hold the control surface against the air loads, and

(*b*) at a certain span depending on the follow-up ratio a negative spring would be needed.

At the point where a zero strength spring is needed to meet the design case, the tabs themselves are just producing sufficient hinge moment to give the pilot the design force (the tabs are assumed here to

(*i*) have a span 100 per cent. of the control span,

(*ii*) have a b_3* equal to -0.4, *i.e.* twice the assumed b_2 of -0.2; this is to ensure small tab angles are used, and

(*iii*) be moderately balanced to a c_3 of -0.1).

The effect of change in the follow-up ratio from 3:1 to 1:1 is to increase the wing span at which zero spring strength is needed from 240 ft. to over 300 ft. It must be borne in mind, however, that the design of tabs assumed in this figure, *i.e.* 100 per cent. of the control span, gives very low tab hinge moments and that any reduction in span will lower both the curves and cause a zero stiffness spring to be needed at a lower wing span.

We can draw the conclusion, therefore, that, while conventional spring tabs can be used up to wing spans of 150-200 ft., somewhere in the region of 150 ft. span the spring is becoming so weak relative to the control hinge moments that it might as well be deleted, and the best design of servo tab system used from the outset. In any case, if a spring tab is used for large aircraft of the order of 200 ft. span, it is likely that the follow-up ratio of the system will have to be reduced to a lower figure than the 3:1 now in general use. For aircraft above 250 to 300 ft. span an attempt at a spring tab design will result in a servo tab design for all practical purposes.

* The coefficients used in the text are defined at the end of the paper.

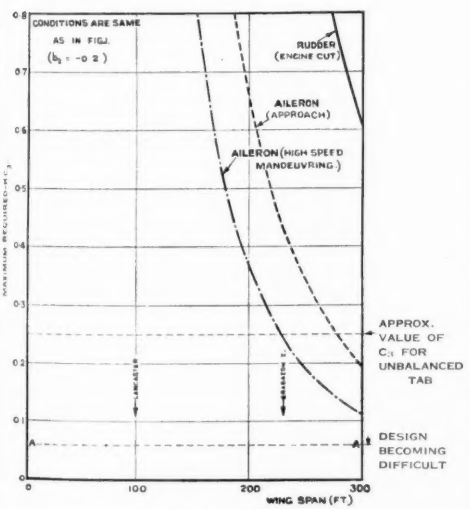

Fig. 3.

Typical variation with size of required Kc_3 of tab to meet various specified flight conditions with pure servo-tab controls (0:1 follow-up ratio).

3. SERVO-TAB CONTROL.

This system of control, which must not be confused with the geared tab system which is a variant of the fixed balance system, is one in which the pilot's control is connected directly only to the tabs on the control surface.

Let us consider first the case with the follow-up ratio equal to zero. Using the same design cases as considered in Fig. 1 for the rudder and aileron controls and assuming

(*a*) the main controls are balanced to a moderate b_2 of -0.2,

(*b*) all gaps are sealed,

(*c*) the tab span is 100 per cent. of the control span (this, in conjunction with (*a*) and (*b*), gives the least adverse control effect of the tab, and also helps to reduce tab inertia to a minimum, an important factor in producing satisfactory flutter characteristics),

(*d*) the b_3 of the tab is twice the b_2 (this ensures sufficient tab power to apply maximum control angles),

we get the position as illustrated in Fig. 3. Here the maximum required Kc_3 for the tab, to meet the force conditions as listed in Fig. 1, is plotted against the wing span of the same "typical" aircraft (Kc_3 for the tab is, of course, equivalent to the Kb_2 of the main control surface). Now the chord of the tab needed to drive the main controls comes out in this case to about 3 per cent. of the total aerofoil chord and the C_3 of an unbalanced tab would be of the order of -0.25. The line AA representing the value of Kc_3 at which design difficulty would become intense is drawn at a value of 0.06; this is lower than that chosen for the Kb_2 because of the much lower hinge moment coefficients of the unbalanced tabs. It is clear that, using conventional control surfaces, on aircraft up to 300 ft. span, there is no problem whatso-

ever in achieving light forces. On the contrary, if the designer wished the tabs to supply all the "feel" force to the pilot, the difficulty would often be in multiplying up the c_3 of the tab to achieve that result. The problem could be tackled by raising the follow-up ratio, for that causes a multiplication of the tab hinge moments that would be felt in the 0:1 case very approximately by the factor

$$\left(1+N\frac{\xi_{\max}}{\beta_{\max}}\right)$$

where N is the follow-up ratio

$\xi_{\max}$ is the maximum control angle

and

$\beta_{\max}$ is the maximum tab angle necessary to drive the control.

In the case considered here $b_3=2b_2$, so that this factor would be approximately equal to $(2N+1)$. There are, however, great objections to the use of increased follow-up ratio; as pointed out in connection with spring tabs, there is the increased difficulty in meeting the flutter criteria. Moreover, it has been pointed out fairly recently that if the present criteria are met with a fair margin with full mass balance, it may be quite practicable to obtain a flutter-free system with the mass balance removed from both tab and *main control surface*. Although this is still under consideration, the possibilities must be kept well in mind, for the present mass balancing penalty in weight on large, long range aircraft is very heavy indeed; if the theoretical investigations continue to be satisfactory we intend to test the results in flight.

To return to the question of feel, there is another and possibly more attractive alternative; that is, to make the tab hinge moment contributions to the pilot's forces as small as possible, and to supply the pilot with purely artificial feel. This would be especially attractive if it were found that, in attempting to rely on tab hinge moments only, the feel, although satisfactory from the viewpoint of repeatability, contained unsatisfactory features due to the interaction of the great number of parameters which control the hinge moments of the tabs. Looked at from this viewpoint, it is evident that even up to spans of 300 ft., we would only have minor trouble with the aileron high speed manœuvring case.

It is worth noting that the aileron cases, which were only slightly worse than the elevator and rudder cases for the fixed balance case of Fig. 1, have become many times more severe for the servo tab case. This is because a tab of given chord is equally effective on any main control surface whose chord ratio to the main aerofoil surface is roughly within the range 15 - 60 per cent.; that is, b_3 is independent of the main control chord over this range. Thus, if a tab of 3 per cent. of the total chord moving through 10° is sufficient to move a main control of 20 per cent. chord through 20° when the b_2 is -0.2, the identical tab moving through the same angle will move a 40 per cent. control of the same span through 20°, provided that the b_2 is still -0.2.

Conventional ailerons are, of course, of much less percentage chord than conventional rudders and elevators. This leads at once to reconsideration of the shape of the ailerons which we should use with servo-tabs, for if we compare ailerons of given rolling power it is clear from the above that smaller tabs will be necessary on the wider chord ailerons. The effect is illustrated in Fig. 4, where the maximum required Kc_3 is plotted for three different ailerons:

(*a*) a conventional aileron of 20 per cent. chord,

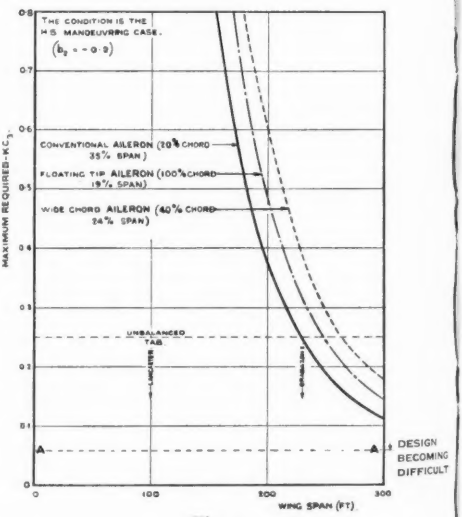

Fig. 4.

Typical variation with size of required Kc_3 for servo tabs (0 : 1 follow up) on three different types of aileron with the same rolling power.

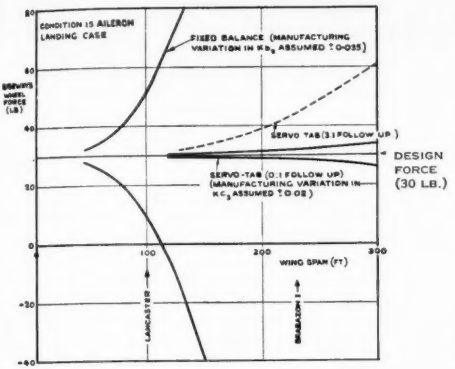
CONDITION IS AILERON LANDING CASE
FIXED BALANCE (MANUFACTURING VARIATION IN KB_2 ASSUMED ±0·035)
SERVO TAB (3:1 FOLLOW UP)
DESIGN FORCE (30 LB.)
SERVO-TAB (0:1 FOLLOW UP) (MANUFACTURING VARIATION IN KC_3 ASSUMED ±0·02)
WING SPAN (FT)
LANCASTER
BRABAZON I

Fig. 5.

Probable scatter of actual wheel forces about the design wheel force for fixed balance and pure servo-tab controls, due to manufacturing variations.

(*b*) a wide chord aileron of 40 per cent chord, and

(*c*) a floating tip aileron of 100 per cent. chord.

It will be seen that, on both the more unconventional ailerons, appreciably less tab balance is required than on the conventional aileron. This result is in complete opposition to the situation on directly operated ailerons, where balance difficulties in addition to inertia difficulties become intense with the wider chord ailerons. It would appear, therefore, that with servo-tab operation of controls, we can re-open the question of fitting wider chord or even floating tip ailerons, with their advantage of enabling wider span landing flaps to be used.

Figure 5 shows graphically the difference in degree between the difficulties of balance design with manually-operated fixed balance controls and those with servo-tab operated controls. Here, the approach case for the aileron has been taken, that is a design case of 30 lb. for full aileron, and the effect of manufacturing tolerances on the design stick load has been evaluated: the effect of manufacturing tolerances on the b_2 has been taken as ± 0.035, and on c_3 as $\pm 0.015 \rightarrow 0.02$. The tremendous advantage of the servo-tab from this particular viewpoint is obvious, even when a large follow-up ratio is used.

Finally, in connection with servo-tabs brief reference will be made in the following paragraphs to some of their particular problems, the more important of which are:—

(*a*) ensuring adequate accuracy of control during take-off and landing runs,

(*b*) ensuring adequate control during stalled conditions,

(*c*) avoiding damage due to the main controls hitting the stops during taxi-ing and in flight,

(*d*) flutter prevention.

3.1. Control during take-off and landing.

There is often some fear that, because the pilot only has positive mechanical control over the tabs, insufficient power will be available from the tabs at very low forward speeds to move the main controls. One should remember that, with the possible exception of the fin in severe cross winds, the aircraft's main surfaces should be unstalled during the ground or water run, and there is, therefore, no reason to doubt that the tab will be producing some moment on the main controls; at any given angle of incidence, the tab hinge moments will bear a given relationship to the overall moments being produced on the aircraft, independent of aircraft speed.

We should expect, therefore, that the tabs would be able to move the controls as soon as the controls can produce enough change in moment on the aircraft to affect its path, provided that there is not excessive frictional resistance to the main control surface movements. This last proviso is most important, as it not only affects the minimum speed at which the main control can be moved at all, but also, what is possibly more vital, affects the accuracy of control at take-off speed; if this accuracy is too poor, the pilot will inevitably over-correct because of the jumpy action of the main control. Thus every precaution has to be taken to keep the friction down to a low limit. As only the main control surface bearings are concerned, normally this is a simple task; but it is clear that cross connecting circuits between two parts of the same control, say between two ailerons, should be avoided, or else extreme care taken with regard to friction of this circuit.

With regard to the possible stalling of the fin during the take-off in a severe cross-wind, this is only a condition which can occur during the early stages of take-off on a runway, and rough control can probably be maintained with brake and throttle until sufficient air speed has been attained over the

rudder by a combination of slipstream and forward speed to unstall the fin; in any case, the power of the main control surface itself is dubious under such conditions. The best solution is to fit a tricycle undercarriage, as then control is hardly required until the take-off point is reached.

3.2. Control during stalled conditions.

It must be admitted that there will be considerable doubt as to the power of the tab when the surface to which its main control is attached is stalled. However, the design must ensure that stalling of both the fin and tail plane will be avoided irrespective of whether there is servo-tab control or any other form of control. We must accept stalled conditions on the wing and the probable loss of, or serious reduction in, tab power on the ailerons as the wing tip stall occurs. Fortunately, loss of, or severe reduction in, aileron power at the stall is not serious, as the elevator is the all-important control in that condition.

3.3. Prevention of damage to the main control surface stops.

There are two distinct conditions which must be considered here, firstly the threshing about of the main control surfaces in the wind when the aircraft is at a standstill or taxi-ing slowly on the ground, and secondly the over-shooting of the control beyond its maximum desired equilibrium position in flight.

Three methods have been suggested to overcome the first condition:—

(*a*) the provision of remotely-controlled locking devices on the main surfaces,

(*b*) the provision of hydraulic dampers between the main control surfaces and the fixed surface,

(*c*) the provision of elastic stops.

The first suggestion is positive, but it has one serious drawback, the possibility of the locks being left in for flight either by negligence, or if inter-connection between, say, the throttles and the locks is provided, by the failure of the withdrawal mechanism. The danger is very real because the locking may not be apparent to the pilot during the normal testing of the cockpit controls before flight. The objections to the second suggestion are that the damping of the control will be felt

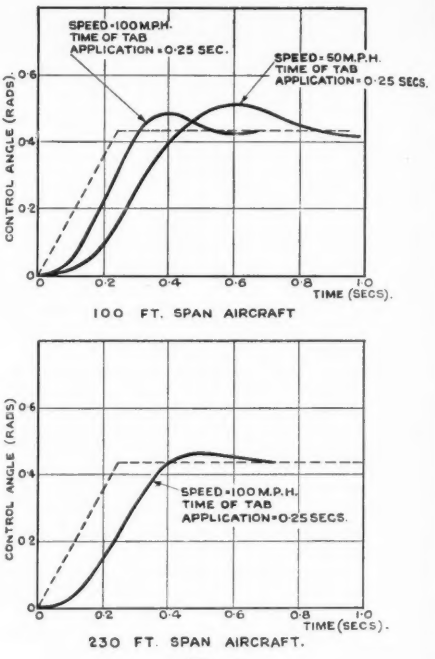

Fig. 6.

Oscillation of servo-tab controls.

in flight as well as on the ground, and that the friction on the main control hinge is bound to rise because of the damper, both to the detriment of the feel in the air. The third suggestion, the provision of elastic stops, seems to us a practical one, and we are at present trying this out in flight at the R.A.E. on a Lancaster fitted with servo-tab controls. The stops in this case are rubber.

Some calculations have been made to investigate the seriousness of the over-shooting of the control in flight. Some of the results of these calculations are shown in Fig. 6. Here the beneficial effect of both increase in speed and size is shown. We can tackle this sort of problem in a number of ways, for example, by providing clearance so that the main controls can over-shoot, or by restricting the rate of tab movement, but as the energy that has to be absorbed at the equilibrium position in flight, in general, is much less than that which has to be absorbed in the ground case, it would appear reasonable to use the elastic stops to prevent the control overswinging past the maximum steady deflection in this case also.

3.4. FLUTTER PREVENTION.

Attention has already been drawn to the possibility of the building of flutter-free servo-tab systems, without mass balance on the tab or main controls. It is only fair to add that the problem of preventing flutter of all kinds on servo-tab controls, an exactly similar problem as for spring tab controls, is not completely solved to everybody's satisfaction. On the other hand, we are fairly close to a complete understanding of the methods of flutter prevention, and we are building up a vast experience with trouble-free systems, and with methods of curing those which have encountered trouble of this kind. Fortunately, because of the development of certain criteria these troubles are not likely to be catastrophic nowadays.

4. POWERED FLYING CONTROLS.

The first essential fact with powered controls is that their use removes any difficulty with control surface balance; any control of any size can be operated at any speed by the pilot, provided that enough amplification of his power is available from the engines or air stream. Thus we have no difficulty in ensuring that the design control forces can be achieved under steady conditions, so long as we start to design a powered control layout from scratch and are not hampered by existing designs of control circuits. The second fact is that there is no fundamental difference between a "power-operated" control and a "power-assisted or boosted" control. The "power-assisted" control is merely one form of powered flying control in which the pilot's feel is given to him as a proportion of the load the power unit is holding. Alternatively, the pilot can be given artificial feel as with the servo-tab installation.

What we have really done when we leave manually-operated controls of any kind, and go to a powered flying control, is to exchange an aerodynamic problem of balance for the mechanical problems of producing a stable, non-oscillating power system with negligible parasitic forces to operate it, and of ensuring extremely good reliability under all assumed forms of failure. The first problem, although difficult with some forms of power systems—notably valving controls—is amenable to mechanical development fairly readily; but the second is far reaching, because we must have exceptionally high standards of reliability for aircraft controls, for complete failure of any one control is almost certain to lead to catastrophe.

There are two distinct ways of treating this second problem:—

(*a*) to use a simple power system and, if failure occurs, to revert to direct manual control, perhaps only with limited control power, and

(*b*) to duplicate all parts of the power system in such a way, that complete failure of any one control is made extremely unlikely.

There is a limit to the size of the aircraft to which method (*a*) is applicable, as shown in Fig. 7. Here the maximum required Kb_2 to achieve certain design conditions with fixed balance control is again plotted against aircraft wing span; the design conditions are based on the lowest control movements which are considered necessary to make an emergency landing and the highest forces which the pilot can stand for a short time. The design of all three controls to meet this emergency case becomes difficult at about 150 ft. span, if there is no change of gearing made between the pilot's control and the control surface as the emergency reversion to manual control is selected; with the highest reasonable change in gearing, say about 3:1, this limit rises to about 200 ft. span. Beyond 220 ft. span or so, the whole problem becomes impossibly difficult unless some aerodynamic servo is used, such as a spring or servo-tab; but if the aerodynamic servo has to be designed as a standby it might as well be done a little more exactly and used as the main method of control and the powered control abandoned.

Moreover, there is one form of failure of the power system that needs to be dealt with specially, even if a manual control in emergency is practical; that is the runaway control. A manually-operated switchover to manual control is not acceptable if this risk is taken into account, as the damage may be done to the aircraft before the pilot has time to operate the reversion switch. If this is at all likely to occur, therefore, we should incorporate a reliable automatic reversion to manual control to deal with failure.

The duplication of the power system to prevent failure, the system which must almost certainly be used if powered controls are fitted to the very large aircraft (wing span > 200 ft.), is extremely difficult. If we are to satisfy ourselves that the drive to a

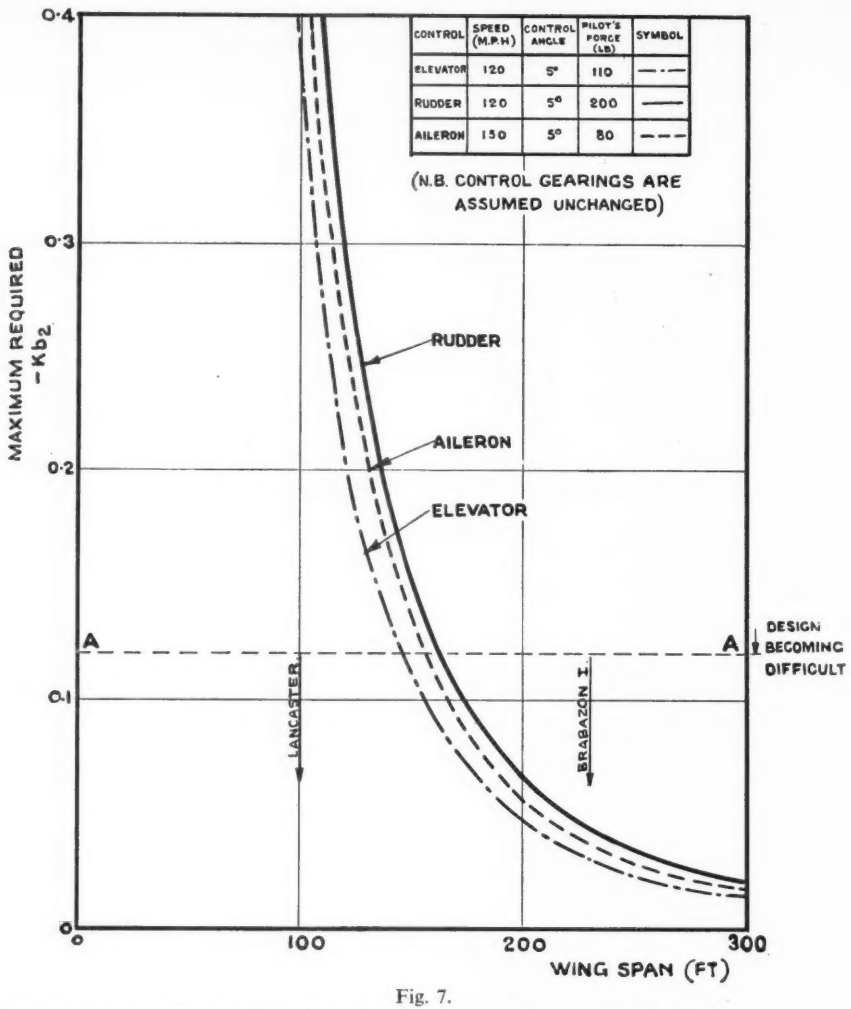

Fig. 7.

Typical variation with size of required Kb_2 of a powered flying control for which the emergency system to effect a landing is reversion to manual control.

control is made sufficiently reliable, we must ask not for a system which "fails safely," but for a system which will operate continuously despite any failure that is recognised as possible or probable, including failures in the reversion mechanisms. It might be better to subdivide the total control surface acting about each of the three axes of the aircraft into three or four completely separate sections and to drive all these sections by remote (say, electrical) control from the pilot's cockpit. Then in case of failure, any one section would be completely isolated without any detection apparatus or switch-over mechanisms. The aircraft would have to be stressed for the condition in which one section of each control surface was locked fully over. There would be a loss of control power in emergency depending on the number of sections (*i.e.* with four sections, up to one half the total control power).

One of the advantages which is obtained with a powered flying control without manual reversion and without "feedback" of feel, is

the possibility of the removal of mass balance weights; but there are certain limits on the permissible backlash and elasticity of the drive between the fixed surface and the control. Further, it is found that, when compared with a fully mass-balanced manually-operated system such as the servo-tab system, a weight more than that saved in mass balance might well be swallowed up in the power system with all its duplication. In addition, as pointed out before, there is the possibility that the mass balance weights of servo-tab systems can be deleted.

5. PILOT'S "FEEL" (*i.e.* The Control Forces Supplied to the Pilot).

In previous sections, reference has been made to the provision of artificial feel for the pilots of large aircraft, as an attractive alternative to giving the pilot the true feel of the balanced hinge moments of the actual controls, or the tab-hinge moments. There are a number of reasons for this suggestion, probably the most important being:—

(*a*) with powered controls the "feel" will be made independent of any control changes, and the stability will generally correspond to the stick-fixed stability of the basic aircraft, *i.e.* the effects of changes in b_1 and b_2 are absent. With servo-tab controls, the effect of changes in c_1, c_2 and c_3 will be very small, although the stability will generally correspond to the stick-free stability, taking account of the fact that follow-up increases the effective b_2 of the main surface.

(*b*) the "feel" can be changed extremely easily by alteration in spring strength, hydraulic pressure or whatever is used to give the artificial feel; if convenient this can be done in flight during the prototype testing.

(*c*) by placing the feel unit close to the pilot the friction in the control circuit can be appreciably reduced, and

(*d*) we have the opportunity of varying the control forces with flight speed or other quantities in any desired manner.

It is possible that on a civil aircraft where there is little danger of the pilot mishandling the aircraft at high speed and there is little need, therefore, to provide stick force warnings of the approach of dangerous stressing conditions, plain spring feel independent of speed will suffice. The rates of such springs would then be unlikely to be such as to give forces exceeding those assumed in Fig. 1, *i.e.* about 5 lb./degree for elevator and rudder and 2 lb./degree for the aileron; we are at present flying an aircraft with forces approximately half of these figures. On military aircraft, where it is essential that warnings must be given of high stresses being imposed, plain spring feel will obviously not do. Here we could make our artificial feel correspond with natural feel, that is to vary the forces as the square of the flight speed. But there is an objection to this and, in fact, to all aerodynamic feel on large military aircraft, except perhaps on the ailerons. It is the difficulty of making a compromise between reasonable manœuvring forces at high speed and at landing, and ensuring high forces as high stressing conditions arise. The worst case, that for the elevator, is illustrated in Fig. 8.

Here is shown the relationship between the maximum loads the pilot can exert and those

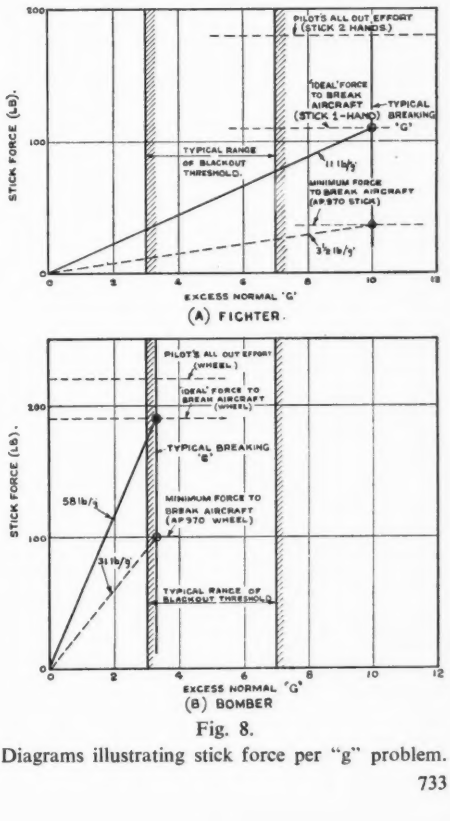

Fig. 8.

Diagrams illustrating stick force per "g" problem.

necessary to break the aircraft. Fig. 8A shows the condition for a fighter aircraft; although we allow an extremely low load of about 35 lb. to break the aircraft we are guarded to a considerable extent by the fact that the blackout threshold of the pilot occurs, except in extremely rare cases, well away from the breaking "g." On bomber aircraft, Fig. 8B, there is practically no safeguard to be had from the blackout threshold, but from manœuvring considerations we have to lower the minimum stick force to break the aircraft, from the 190 lb. or so we would probably like, to 100 lb. Even so the minimum stick force per "g" allowed is of the order of 30 lb./g, *i.e.* about 10 times that for the fighter, although we have the same man manœuvring both types. Basing an estimate on a constant force to manœuvre the aircraft to the desired limit of normal "g" corresponding to the duty, we should like the stick force per "g" to be about three times as high on the bomber as on the fighter.

A solution to this problem has been thought for some time to be the"g" restrictor, of which the desired characteristics of one type are illustrated in Fig. 9. These are based on the non-interference of the "g" restrictor at low normal accelerations in the normal manœuvring range with a rapidly increasing stick force being added into the circuit load by the "g" restrictor as the "g" approaches the limiting value. "G" restrictors of this type, both aerodynamic and mechanical, have been made and have been, or are being, tested; the aerodynamic solution is not now considered safe as it seriously increases the flutter risk. There are other possible types which operate by restricting stick movement but they are not so convenient for meeting the military requirements. The mechanical type can be made to fit in conveniently with the artificial feel generator, which due to the fitting of the "g" restrictor, may not be required to vary with speed; feel which does not vary with speed may not, however, be perfect as, unless special care is taken with the friction in the circuit, the control will become increasingly sensitive with increase

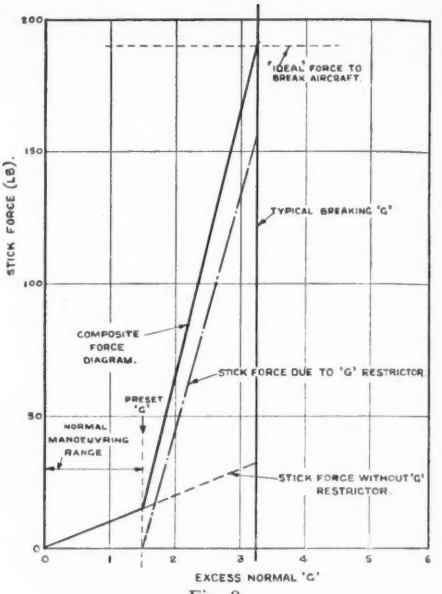

Fig. 9.

Diagram illustrating characteristics of stick force type "g" restrictor for a bomber aircraft.

in flight speed, because of the smaller control movements necessary at the higher speeds.

DEFINITIONS.

b_1 is the rate of change of *control* hinge moment coefficient with *incidence* of the main surface.

b_2 is the rate of change of *control* hinge moment coefficient with *control* angle.

b_3 is the rate of change of *control* hinge moment coefficient with *tab* angle.

c_1 is the rate of change of *tab* hinge moment coefficient with *incidence* at the main surface.

c_2 is the rate of change of *tab* hinge moment coefficient with *control* angle.

c_3 is the rate of change of *tab* hinge moment coefficient with *tab* angle.

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ROTARY DERIVATIVES OF A DELTA WING AT SUPERSONIC SPEEDS

by

A. ROBINSON, M.Sc., A.F.R.Ae.S.

SUMMARY

The rotary derivatives of a Delta wing pitching or rolling at supersonic speed are obtained in addition to the lift curve slope first given in an earlier investigation.

1. GENERAL.

1.1. INTRODUCTION.

In the present paper, we obtain on the assumptions of linearised theory, formulæ for the rotary derivatives of a Delta wing pitching or rolling at supersonic speed. The results are arrived at by an extension of a method used in earlier papers to find the lift curve slope of a Delta wing, and values of the latter are included in the present paper for the sake of completeness. It will be found appropriate throughout to distinguish between two cases according as the apex semi-angle of the aerofoil is (*i*) smaller, or (*ii*) greater, than the Mach angle, and different methods will be used to deal with the two cases.

This distinction is more than a matter of mere analytical convenience. In fact, it will appear that case (*i*), which is the condition first reached by a given aerofoil on entering supersonic regions, has certain features in common with subsonic conditions, which are not shared by case (*ii*), corresponding to even higher speeds. In consequence, case (*i*) in the sequel will be referred to as the "quasi-subsonic" case, while case (*ii*) will be characterised as "definitely supersonic."

1.2. NOTATION.

ρ = air density.
p = pressure.
Δp = excess pressure.
$\bar{\Delta p}$ = pressure difference between top and bottom surface.
V = free stream velocity.
a = speed of sound.
M = Mach number.
μ = Mach angle.
$\beta = (M^2 - 1)^{\frac{1}{2}} = \cot \mu$.
S = surface area of Delta wing.
b = span.
c = maximum chord.
$\bar{c}$ = geometrical mean chord $\left(= \frac{c}{2} \right)$
A = aspect ratio
γ = apex semi-angle.
x = chordwise co-ordinate (measured from the apex against the direction of flow).

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y = spanwise co-ordinate (positive to starboard).
z = normal co-ordinate.
α = incidence.
p = angular velocity in roll.
q = angular velocity in pitch.
L = lift.
$\bar{L}$ = rolling moment.
$\bar{M}$ = pitching moment.
C_L = lift coefficient.
l_p = non-dimensional derivative of rolling moment with rate of roll $= 4(\partial\bar{L}/\partial p)/\rho V S b^2$.
m_q = non-dimensional derivative of pitching moment with rate of pitch $= (\partial\bar{M}/\partial q)/\rho V S c^2$.

$K(u)$ = complete elliptic integral of the first kind $= \int_0^{\pi/2} \frac{d\phi}{(1-u^2\sin^2\phi)^{\frac{1}{2}}}$

$K'(u)$ = complementary complete elliptic integral of the first kind $= K[(1-u^2)^{\frac{1}{2}}]$.

$E(u)$ = complete elliptic integral of the second kind $\int_0^{\pi/2} (1-u^2\sin^2\phi)^{\frac{1}{2}} d\phi$

$E'(u)$ = complementary complete elliptic integral of the second kind $= E[(1-\mu^2)^{\frac{1}{2}}]$.

1.3. Results.

Our results are summarised below:—

The lift curve slope is given by

$$\partial C_L/\partial\alpha = -(2\pi\tan\gamma)/E'(\cot\bar{\mu}\tan\gamma) \quad . \quad . \quad . \quad (1, i)$$

in case (*i*), γ not less than $\bar{\mu}$.

$$\partial C_L/\partial\gamma = -4\tan\bar{\mu} \quad . \quad . \quad . \quad . \quad . \quad . \quad (1, ii)$$

in case (*ii*), γ not greater than $\bar{\mu}$ and by

The damping derivative of rolling moment with the rate of roll is given by

$$l_p = -\frac{\pi}{4}\frac{(\tan^2\bar{\mu}-\tan^2\gamma)\tan\gamma}{(2\tan^2\bar{\mu}-\tan^2\gamma)E'(\cot\bar{\mu}\tan\gamma)-\tan^2\gamma K'(\cot\bar{\mu}\tan\gamma)} \quad . \quad (2, i)$$

in case (*i*)

and by

$$l_p = -\frac{1}{3}\tan\bar{\mu} \quad . \quad . \quad . \quad . \quad . \quad . \quad (2, ii)$$

in case (*ii*).

The damping derivative of pitching moment with the rate of pitch is given by

$$m_q = -\frac{3\pi(\tan^2\bar{\mu}-\tan^2\gamma)\tan\gamma}{(\tan^2\bar{\mu}-2\tan^2\gamma)E'(\cot\bar{\mu}\tan\gamma)+\tan^2\gamma K'(\cot\bar{\mu}\tan\gamma)} \quad . \quad (3, i)$$

in case (*i*)

and by

$$m_q = -4\tan\bar{\mu} \quad . \quad . \quad . \quad . \quad . \quad . \quad (3, ii)$$

in case (*ii*)

In the above formulæ, it is assumed that the wing pitches round an axis through the apex, and that the damping moment is taken round the same axis. The corresponding total lift on the aerofoil can be calculated from

$$\frac{\partial C_L}{\partial (\bar{c}q/V)} = -\frac{4\pi(\tan^2\bar{\mu} - \tan^2\gamma)\tan\gamma}{(\tan^2\bar{\mu} - 2\tan^2\gamma)E'(\cot\bar{\mu}\tan\gamma) + \tan^2\gamma K'(\cot\bar{\mu}\tan\gamma)} \qquad (4, i)$$

in case (*i*)

and from

$$\frac{\partial C_L}{\partial (\bar{c}q/V)} = -(16/3)\tan\bar{\mu} \qquad . \quad . \quad . \quad . \quad (4, ii)$$

in case (*ii*)

$\partial C_L/\partial\alpha$, l_p, m_q and $\partial C_L/\partial(\bar{c}q/V)$ are plotted against Mach number for various apex angles (or aspect ratios) in Figs. 1 to 4, respectively.

Given an aerofoil, we might estimate the forces acting on it by assuming that the pressure difference on any surface element is proportional to the local incidence, as in Ackeret's two-dimensional theory. In analogy with the corresponding procedure for low speed flow, this will be called the "strip-theory method." Fig. 5 shows the ratio of the values of $\partial C_L/\partial\alpha$ etc., as obtained by "exact" linearised theory, and of the corresponding values according to strip-theory. It will be seen that these ratios not only tend to one as γ tends to 90°, as might be expected, but are all actually equal to one as soon as $\gamma \geqslant \bar{\mu}$.

The chordwise position of the centre of pressure due to incidence is at a distance of $4/3\ \bar{c}$ aft of the apex, and the chordwise position of the centre of pressure due to pitching round the apex is at a distance of $3/2\ \bar{c}$ aft of that point. Now, on the

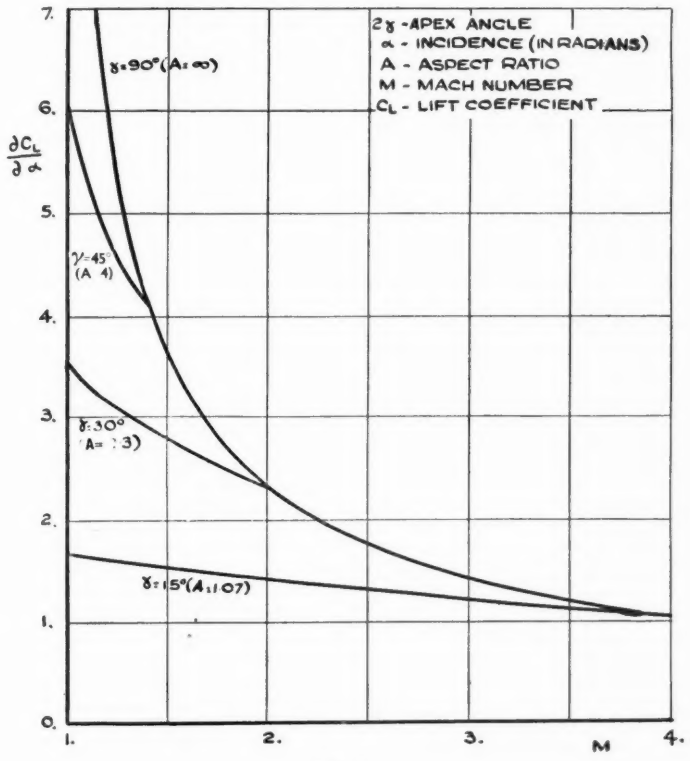

Fig. 1.
Variation of lift curve slope with Mach number for various apex angles.

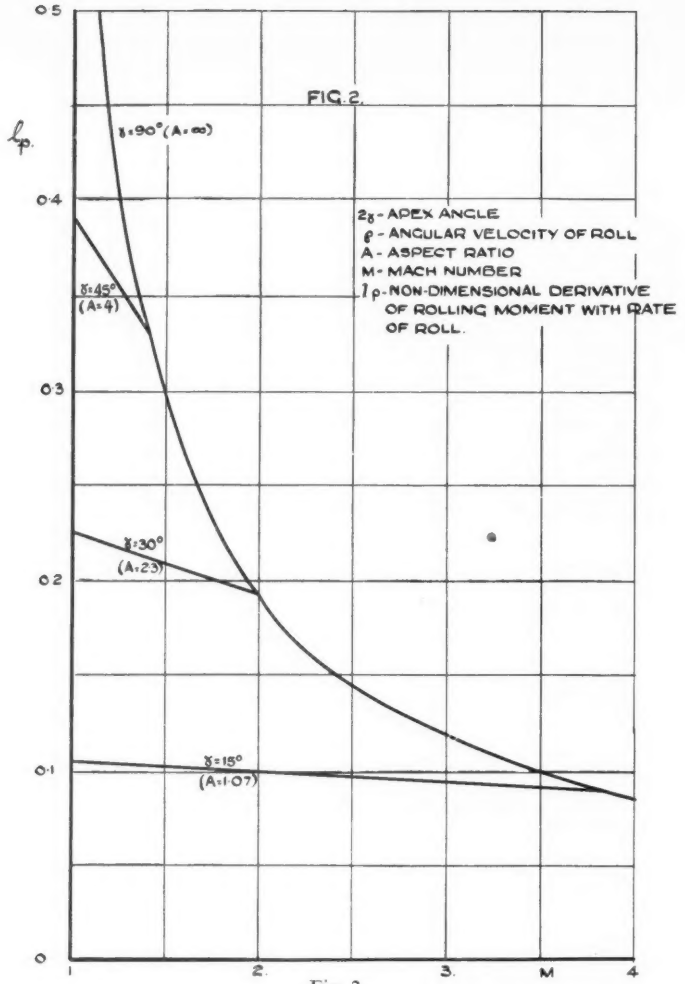

Fig. 2.
Variation of rolling moment derivative with Mach number for various apex angles.

assumptions of linearised theory, vertical motion of the aerofoil with velocity w is equivalent to an incidence w/V. Also, to the same order of accuracy, if the aerofoil pitches at an angular velocity q round an axis which is at a distance d aft of the apex, this amounts to pitching at the same velocity q round the apex, together with vertical motion at a velocity $-dq$. Hence the moment round the apex in that case is

$$[m_q - (2d/3\bar{c})(\partial C_L/\partial\alpha)]\rho V q S\bar{c}^2$$

and the normal force is

$$[\partial C_L/\partial(\bar{c}q/V) - (d/\bar{c})(\partial C_L/\partial\alpha)]\tfrac{1}{2}\rho V q S\bar{c}$$

acting at a point

$$2[m_q - (2d/3\bar{c})(\partial C_L/\partial\alpha)]\bar{c}/[\partial C_L/\partial(\bar{c}q/V) - (d/\bar{c})(\partial C_L/\partial\alpha)]$$

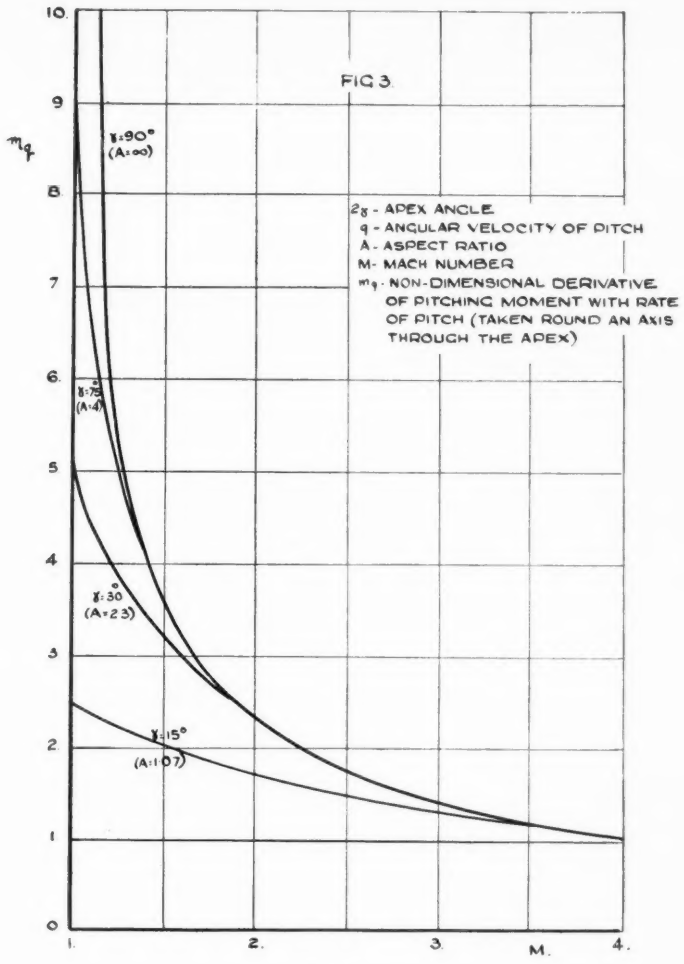
FIG 3.
m_q
2β - APEX ANGLE
q - ANGULAR VELOCITY OF PITCH
A - ASPECT RATIO
M - MACH NUMBER
m_q - NON-DIMENSIONAL DERIVATIVE OF PITCHING MOMENT WITH RATE OF PITCH (TAKEN ROUND AN AXIS THROUGH THE APEX)
2β=90° (A=∞)
2β=75° (A=4)
2β=30° (A=2·3)
2β=15° (A=1·07)
M.

Fig. 3.
Variation of pitching moment derivative with Mach number for various apex angles.

aft of the apex. In the above expressions, m_q, $\partial C_L/\partial\alpha$ and $\partial C_L/\partial(\bar{cq}/V)$ are still given by formulæ (1, *i*)—(4, *ii*), so that these formulæ are in fact adequate for the determination of the damping moment due to pitching round an arbitrary axis.

2. ANALYSIS.

2.1. THE MATHEMATICAL PROBLEM.

Consider a Delta wing (triangular aerofoil) of span b and maximum chord length c in a uniform supersonic air stream (Fig. 6). The linearised equation of steady supersonic flow is (compare Ref. 1)

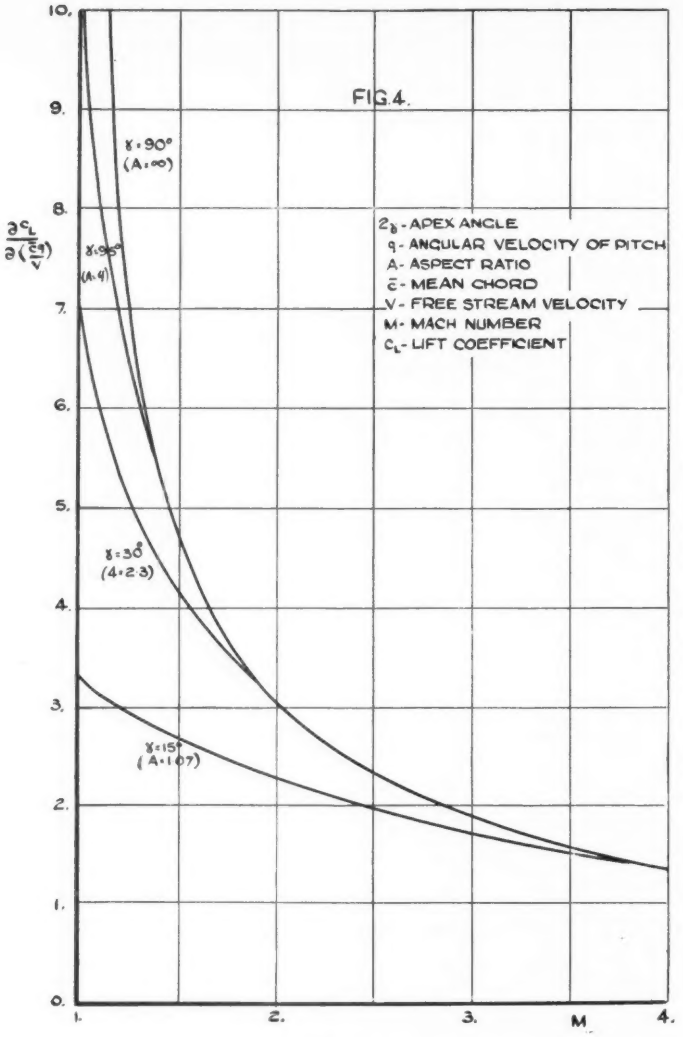

Fig. 4.
Variation of lift due to pitching with Mach number for various apex angles.

$$\beta^2\partial^2\Phi/\partial x^2 - \partial^2\Phi/\partial y^2 - \partial^2\Phi/\partial z^2 = 0 \quad . \quad . \quad . \quad (5)$$

where $\beta^2 = M^2 - 1$, $M = V/a$, M being the Mach number, V the free stream velocity, and a the speed of sound; x is the longitudinal co-ordinate measured from the apex against the direction of flow, y the lateral co-ordinate, positive to starboard and negative to port, and z the normal co-ordinate, positive downward. Φ is the induced velocity potential, so that the three velocity components are given by

$$-V + \partial\Phi/\partial x,\ \partial\Phi/\partial y, \text{ and } \partial\Phi/\partial z$$

respectively.

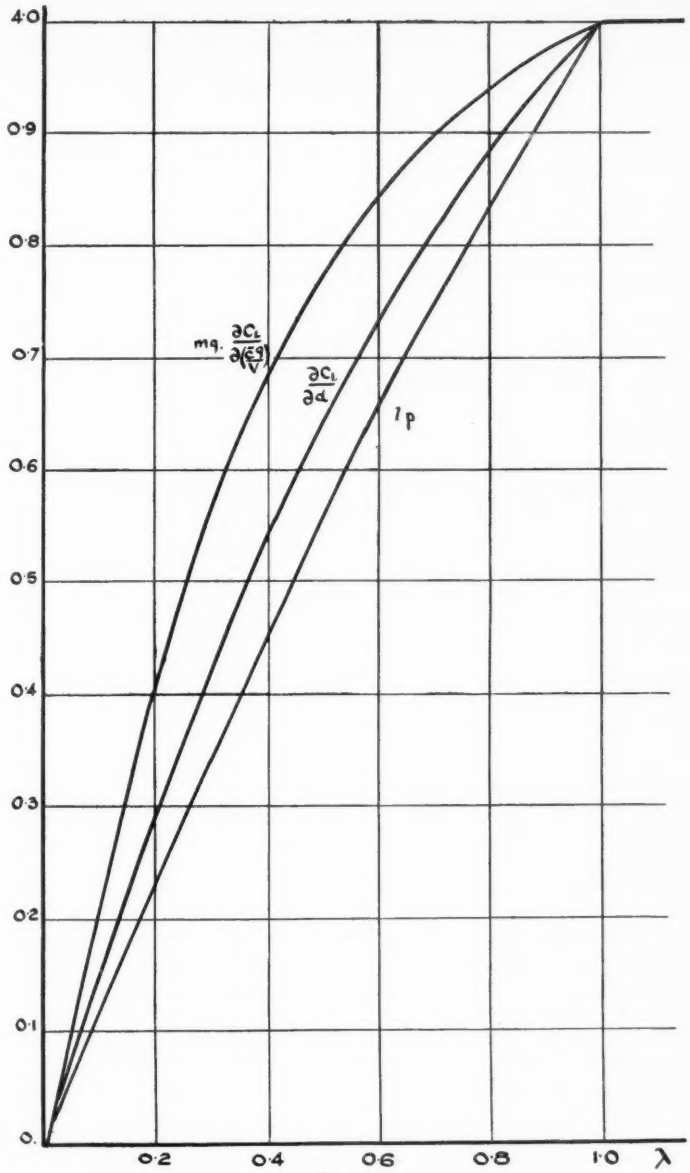

Fig. 5.
(For the explanation, see text.)

In accordance with the conventions of linearised theory, the incidence of the flow at the aerofoil is estimated at the normal projection of the aerofoil into the $x-y$ plane; thus

$$s=(1/V)\,(\partial\Phi/\partial z)_{z=0} \quad . \quad . \quad . \quad . \quad . \quad (6)$$

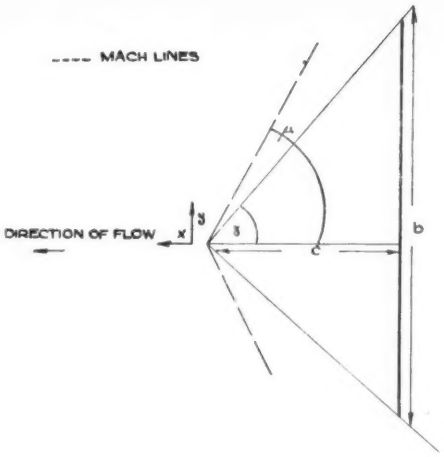

Fig. 6.

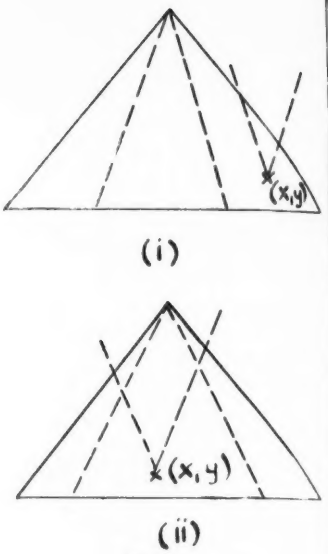

Fig. 7.

where s is the slope of the aerofoil at the point in question, on the upper or lower surface, as the case may be. Φ must be continuous everywhere except possibly across the aerofoil or across its wake.

The longitudinal component of induced velocity is $\partial\Phi/\partial x$ hence by the linearised Bernoulli equation,

$$\bar{p}=\bar{p}_{\infty}-\rho V\partial\Phi/\partial x \quad . \quad . \quad . \quad . \quad . \quad (7)$$

where $\bar{p}$ is the pressure at the point in question, $\bar{p}_{\infty}$ the free stream pressure, and ρ the air density. The excess pressure is therefore given by

$$\Delta\bar{p}=-\rho V\partial\Phi/\partial x \quad . \quad . \quad . \quad . \quad . \quad (8)$$

For a flat aerofoil at a small incidence α, the slope s is given by $s=\tan\alpha\fallingdotseq\alpha$, so that condition (6) becomes

$$\alpha=(1/V)\,(\partial\Phi/\partial z)_{z=\pm 0} \quad . \quad .. \quad . \quad . \quad . \quad (9)$$

If the aerofoil is moving with a small normal velocity w, then its "equivalent slope" is w/v, so that $w=(\partial\Phi/\partial z)_{z=\pm 0}$. The condition is still genuinely steady, and the free stream velocity is still given by V, since $w\leqslant V$, by assumption.

The conditions of "steady" pitch and "steady" roll are not genuinely steady from the kinematic point of view. For low speeds it is customary to simplify these conditions by the assumption that the flow is, in fact, steady, while the local incidence varies linearly in spanwise, or in chordwise, direction as the case may be (see *e.g.* ref. 2). A closer investigation tends to show that this assumption is still admissible in the present case, as long as the product of the angular velocity in question, and of a typical linear dimension of the aerofoil is small compared with the speed of sound. With this reservation, the boundary condition for an aerofoil rolling with angular velocity p becomes

$$py=(\partial\Phi/\partial z)_{z=\pm 0} \quad . \quad . \quad . \quad . \quad . \quad (10)$$

while the boundary condition for an aerofoil pitching round the apex with angular velocity q is

$$qx=(\partial\Phi/\partial z)_{z=\pm 0} \quad . \quad . \quad . \quad . \quad . \quad (11)$$

It should be observed that as long as we confine our attention to the region ahead of the trailing edge, conditions aft of the trailing edge do not affect the results, so that we may modify them at our convenience. In particular, we shall assume that Φ satisfies conditions (9), (10) or (11) everywhere in the triangular region $z=0$. $x<0$, $x^2-y^2\cot\mu>0$ and that it may be discontinuous across that region.

2.2. THE QUASI-SUBSONIC CASE.

Taking first the quasi-subsonic case, we have $\gamma<\mu$ so that the aerofoil is completely inside the Mach cone issuing from the apex. It follows that Φ is a constant, and may be assumed to vanish, outside the cone. In particular, this yields the conditions

$$\Phi=0 \text{ for } x^2-\beta^2(y^2+z^2)=0 \quad . \quad . \quad . \quad . \quad . \quad (12)$$

We now transform x, y, z into hyberboloido-conal co-ordinates r, μ, ν (ref. 3) by means of the equations.

$$x=-r\beta\mu\nu/hk,\ y=rAA_1/hA_2,\ z=rBB_1/kA_2 \quad . \quad . \quad . \quad . \quad (13)$$

where $A=(\mu^2-h^2)^{\frac{1}{2}}$, $A_1=(\nu^2-h^2)^{\frac{1}{2}}$, $A_2=(k^2-h^2)^{\frac{1}{2}}$, $B=(\mu^2-k^2)^{\frac{1}{2}}$ and $B_1=(k^2-\nu^2)^{\frac{1}{2}}$ and $\beta=\cot\mu$, as before, $k=\cot\gamma$, and $h=(\cot^2\gamma-\cot^2\mu)^{\frac{1}{2}}$ so that $k>h$. The intervals of variation of r, μ, ν will be taken as

$$0\leqslant r<\infty,\ k\leqslant\mu<\infty,\ h\leqslant\nu\leqslant k.$$

Eliminating first μ and ν, then r and ν and finally r and μ from (7), we obtain three families of surfaces respectively, namely,

$$x^2-\beta^2(y^2+z^2)=\beta^2r^2 \quad . \quad . \quad . \quad . \quad . \quad (14)$$

$$\frac{x^2}{\beta^2\mu^2}-\frac{y^2}{A^2}-\frac{z^2}{B^2}=0 \quad . \quad . \quad . \quad . \quad . \quad (15)$$

and

$$\frac{x^2}{\beta^2\nu^2}-\frac{y^2}{A_1^2}+\frac{z^2}{B_1^2}=0 \quad . \quad . \quad . \quad . \quad . \quad (16)$$

(14) represents a family of hyperboloids of two sheets, while (15) and (16) are families of cones. To every point satisfying $x<0$, $x^2-\beta^2(y^2+z^2)>0$ there corresponds exactly one triplet r, μ, ν, within the domain of variation of those variables. On the other hand, to each triplet, r, μ, ν, there correspond four points in the x, y, z space, according to the determination of the square roots in (13). The ambiguity can be avoided by writing μ and ν as elliptic functions of new variables, but this procedure will not be required in the present report.

As $\mu\to\infty$, the cones of the family (15) approximate the cone $x^2-\beta^2(y^2+z^2)=0$ while for $\mu\to k$ they tend to become equal to the (two-sided) region in the x - y plane given by $(x^2/\beta^2k^2)-(y^2/k^2-h^2)\geqslant 0$, *i.e.*, $x^2-y^2\cot^2\gamma\geqslant 0$. It will be seen that the vertical projection of the Delta wing coincides with part of that region as specified by $0>x>-c$. The cones of (16) aproximate the complementary angular region in the x -y plane, $(x^2/\beta^2k^2)-(y^2/k^2-h^2)<0$ as $\nu\to k$, and the y-axis ($x=0$, $z=0$), as $\nu\to h$. Thus, the intersections of the μ-cones with the plane $\beta x=1$ are ellipses varying between the circle $y^2+r^2=1$ and the slit $z=0$, $y^2<1-(h^2/k^2)$. The intersections of the ν-cones with the same plane are hyperbolæ.

In the sequel, we shall require the following relations.

$$\left.\begin{aligned}&\partial x/\partial r=-\beta\mu\nu/hk,\ \partial x/\partial\mu=-r\beta\nu/hk,\partial x/\partial\nu=-r\beta\mu/hk\\&\partial y/\partial r=AA_1/hA_2,\ \partial y/\partial\mu=r\mu A_1/hA_2A,\ \partial y/\partial\nu=r\nu A/hA_2A_1\\&\partial z/\partial r=BB_1/kA_2,\ \partial z/\partial\mu=r\mu B_1/kA_2B,\ \partial z/\partial\nu=-r\nu B/kA_2B_1\end{aligned}\right\} \quad . \quad (17)$$

$$\left.\begin{aligned}&\partial r/\partial x=-\mu\nu/\beta hk,\ \partial\mu/\partial x=\nu A^2B^2/\beta hkrB_2^2,\ \partial\nu/\partial x=\mu A_1^2B_1^2/\beta hkrB_2^2\\&\partial r/\partial z=-B^2B_1^2/kA_2^2,\ \partial\mu/\partial z=\mu A^2BB_1/kA_2rB_2^2,\ \partial\nu/\partial z=-\nu A_1^2BB_1/kA_2r_2^2\end{aligned}\right\} \quad (18)$$

where $B_2=(\mu^2-\nu^2)^{\frac{1}{2}}$, and A, A_1, A_2, B and B_1 are as before

and

$$\partial\Phi/\partial x = -(1/\beta hk)\,[\mu\nu\partial\Phi/\partial r - (\nu A^2B^2/rB_2^2)\,(\partial\Phi/\partial\mu) - (\mu A_1^2B_1^2/rB_2^2)\,(\partial\Phi/\partial\nu)] \quad (19)$$

$$\partial\Phi/\partial z = (BB_1/kA_2)\,[-\partial\Phi/\partial r + (\mu A^2/rB_2^2)(\partial\Phi/\partial\mu) - (\nu A_1^2/rB_2^2)\,(\partial\Phi/\partial\nu)] \quad (20)$$

$$\beta^2\partial^2\Phi/\partial x^2 - \partial^2\Phi/\partial y^2 - \partial^2\Phi/\partial z^2 = (1/r^2)\Big[\frac{\partial}{\partial r}\,(r^2\partial\Phi/\partial r) - (AB/B_2^2)$$

$$\frac{\partial}{\partial\mu}\,(AB\partial\Phi/\partial\mu) - (A_1B_1/B_2^2)\,\frac{\partial}{\partial\nu}(A_1B_1\partial\Phi/\partial\nu)\Big]. \quad . \quad . \quad (21)$$

where Φ is an arbitrary scalar point function.

Equations (17) are obtained directly from equations (13), and equations (19) and (20) from equations (18). The labour involved in the direct calculation of equations (18) and (21) can be avoided by an adaptation of the methods used in the theory of orthogonal co-ordinates (see ref. 3).

Now let Φ be a scalar point function satisfying (5). By (21) this is equivalent to

$$B_2^2\,\frac{\partial}{\partial r}\,(r^2\partial\Phi/\partial r) - AB\,\frac{\partial}{\partial\mu}\,(AB\partial\Phi/\partial\mu) - A_1B_1\,\frac{\partial}{\partial\nu}\,(A_1B_1\partial\Phi/\partial\nu) = 0. \quad . \quad (22)$$

Let us try to find solutions of the form $\Phi = r^n\Psi$ where Ψ is a function of μ and ν only. Substitution in (22) shows that the condition for Ψ is

$$n\,(n+1)\,B_2^2\Psi - AB\,\frac{\partial}{\partial\mu}\,(AB\partial\Psi/\partial\mu) - A_1B_1\,\frac{\partial}{\partial\nu}\,(A_1B_1\partial\Psi/\partial\nu) = 0 \quad . \quad (23)$$

Or assuming that Ψ is of the form $\Psi = G(\mu)\,H(\nu)$

$$H(\nu)\,[n\,(n+1)\,\mu^2G(\mu) - \mu\,(A^2+B^2)\,(dG/d\mu) - A^2B^2d^2\,(G/d\mu^2)]$$

$$-\,G(\mu)\,[n\,(n+1)\,\nu^2H(\nu) - \nu\,(A_1^2 - B_1^2)\,(dH/d\nu) + A_1^2B_1^2\,(d^2H/d\nu^2)] = 0 \quad (24)$$

Hence

$$[1/G(\mu)]\,[n\,(n+1)\,\mu^2G(\mu) - \mu\,(A^2+B^2)\,(dG/d\mu) - A^2B^2\,(d^2G/d\mu^2)]$$

$$= [1/H(\nu)]\,[n\,(n+1)\,\nu^2H(\nu) - \nu\,(A_1^2+B_1^2)\,(dH/d\nu) + A_1^2B_1^2\,(d^2H/d\nu^2)]$$

$$= \text{const.} = j\,(h^2+k^2) \quad . \quad . \quad . \quad . \quad . \quad (25)$$

where j is an arbitrary constant. It follows that $G(\mu)$ satisfies the differential equation

$$[n\,(n+1)\,\mu^2 - j\,(h^2+k^2)]G(\mu) - \mu\,(A^2+B^2)dG/d\mu - A^2B^2d^2G/d\mu^2 = 0 \quad (26)$$

with an exactly similar equation for $H(\nu)$.

Equation (26) is Lamé's equation (compare ref. 4). For given n, j can be determined in $2n+1$ different ways so that $G(\mu)$ is of one of the following four forms.

$$\left.\begin{aligned} K(\mu) &= a_0\mu^n + a_1\mu^{n-2} + \ldots \\ L(\mu) &= A\,(a_0\mu^{n-1} + a_1\mu^{n-3} + \ldots) \\ M(\mu) &= B\,(a_0\mu^{n-1} + a_1\mu^{n-3} + \ldots) \\ N(\mu) &= AB\,(a_0\mu^{n-2} + a_1\mu^{n-4} + \ldots) \end{aligned}\right\} \quad . \quad . \quad . \quad (27)$$

where the expressions $a_0\mu^n + a_1\mu^{n-2} + \ldots$, $a_0\mu^{n-1} + a_1\mu^{n-3} + \ldots$ and $a_0\mu^{n-2} + a_1\mu^{n-4} + \ldots$ . . . are all polynomials in μ. These solutions are said to be of the first kind; they are denoted by $E_n^m(\mu)$, $m = 1, 2, 3 \ldots (2n+1)$.

Given $E_n^m(\mu)$, a second solution of Lamé's equation for the same n and j is given by

$$F_n^m(\mu) = E_n^m(\mu) \int_\mu^\infty \frac{dt}{E_n^m(t)\,A_3B_3}$$

where $A_3 = (t^2 - h^2)^{\frac{1}{2}}$ and $B_3 = (t^2 - k^2)^{\frac{1}{2}}$.

Thus, the functions

$$\Phi = r^nF_n^m(\mu)\,E_n^m(\nu) \quad . \quad . \quad . \quad . \quad . \quad (28)$$

are normal solutions of (22) and (5). They have the property that $\Phi \to 0$ on approaching the cone $x^2 - \beta^2(y^2+z^2) = 0$. Also they are continuous inside the cone, except possibly across the triangular region $x^2 - y^2\cot\gamma > 0$, $x < 0$, $r = 0$.

Hence, provided it satisfies the requisite boundary conditions, a function Φ defined by (28) inside the cone $x^2-\beta^2(y^2+z^2)=0$, $x<0$ and by $\Phi=0$ elsewhere may serve as a solution to any particular problem related to a triangular aerofoil.

The particular solutions which will be used in the sequel are all based on functions of the type of $M(\mu)$ and $N(\mu)$ mentioned in (27). It can be shown that both these functions are of the form $H(\mu)=BP(\mu)$ where $P(\mu)$ is a function which does not vanish for $\mu\geqslant k$. The corresponding normal solution is then of the form

$$\Phi=Cr^nG(\mu)H(\nu)$$

where C is an arbitrary constant and

$$G(\mu)=H(\mu)\int_\mu^\infty\frac{dt}{[H(t)]^2A_3B_3}=P(\mu)B\int_\mu^\infty\frac{dt}{[P(t)]^2B_3{}^3A_3}$$

We have

$$\int_\mu^\infty\frac{dt}{[P(t)]^2B_3{}^3A_3}=-\left[\frac{1}{t[P(t)]^2A_3B_3}\right]_\mu^\infty+\int_\mu^\infty\frac{d}{dt}\left[\frac{1}{t[P(t)]^2A_3}\right]\frac{dt}{B_3}$$

$$=\frac{1}{\mu[P(\mu)]^2AB}+\int_\mu^\infty\frac{d}{dt}\left[\frac{1}{t[P(t)]^2A_3}\right]\frac{dt}{B_3}$$

As $\mu\to k$, the integral on the right hand side tends to a finite limit. Hence

$$\lim_{\mu\to k}G(\mu)=\lim_{\mu\to k}P(\mu)B\frac{1}{\mu[P(\mu)]^2AB}=\frac{1}{kP(k)A_2}$$

and so, at the aerofoil,

$$\lim_{\mu\to k}\Phi=\frac{Cr^n\,H(\nu)}{kP(k)A_2}\quad\ldots\quad(29)$$

To find the longitudinal and normal induced velocities, we apply formulæ (19) and (20) and obtain

$$\frac{\partial\Phi}{\partial x}=-C\frac{r^{n-1}}{\beta hk}\left[n\mu\nu G(\mu)H(\nu)-\frac{\nu A^2B^2}{B_2{}^2}\cdot\frac{dG}{d\mu}H(\nu)-\frac{\nu A_1{}^2B_1{}^2}{B_2{}^2}G(\mu)\frac{dH}{d\nu}\right]\quad(30)$$

and

$$\frac{\partial\Phi}{\partial z}=C\frac{r^{n-1}BB_1}{kA_2}\left[-nG(\mu)H(\nu)+\frac{\mu A^2}{B_2{}^2}\frac{dG}{d\mu}H(\nu)-\frac{\nu A_1{}^2}{B_2{}^2}G(\mu)\frac{dH}{d\nu}\right].\quad(31)$$

Again,

$$\frac{dG}{d\mu}=\left(BP'(\mu)+\frac{\mu P(\mu)}{B}\right)\int_\mu^\infty\frac{dt}{[P(t)]^2B_3{}^3A_3}-\frac{BP(\mu)}{[P(\mu)]^2B^3A}$$

$$=\left(BP'(\mu)+\frac{\mu P(\mu)}{B}\right)\frac{1}{\mu[P(\mu)]^2BA}+\int_\mu^\infty\frac{d}{dt}\left[\frac{1}{t[P(t)]^2A_3}\right]\frac{dt}{B_3}-\frac{1}{P(\mu)B^2A}$$

$$=\frac{P'(\mu)}{\mu[P(\mu)]^2A}+\left[BP'(\mu)+\frac{\mu P(\mu)}{B}\right]\int_\mu^\infty\frac{d}{dt}\left[\frac{1}{t[P(t)]^2A_3}\right]\frac{dt}{B_3}$$

and so

$$\lim_{\mu\to k}B\frac{dG}{d\mu}=kP(k)\int_k^\infty\frac{d}{dt}\left[\frac{1}{t[P(t)]^2A_3}\right]\frac{dt}{B_3}$$

Hence, at the aerofoil,

$$\lim_{\mu\to k}\frac{\partial\Phi}{\partial x}=-C\frac{r^{n-1}}{\beta hk}[n\nu H(\nu)-A_1{}^2H'(\nu)]/P(k)A_2\quad\ldots\quad(32)$$

and

$$\lim_{\mu\to k}\frac{\partial\Phi}{\partial z}=Cr^{n-1}P(\nu)\,A_2kP(k)\int_k^\infty\frac{d}{dt}\left[\frac{1}{t\,[P(t)]^2A_3}\right]\frac{dt}{B_3}\quad .\quad .\quad (33)$$

Since $A_2kP(k)\int_k^\infty\frac{d}{dt}\left[\frac{1}{t\,[P(t)]^2A_3}\right]\frac{dt}{B_3}$

is constant at the aerofoil, $\frac{\partial\Phi}{\partial z}$ is proportional to $r^{n-1}P(\nu)$. Hence in order to satisfy condition (9), (10) or (11), we have to choose the solution $H(\nu)=B_1P(\nu)$ in such a way that $r^{n-1}P(\nu)$ is a constant, or that it is proportional to y or to x as the case may be. In the first case $r^{n-1}P(\nu)$ is therefore independent of a linear dimension, so that $n=1$, while in the other cases $r^{n-1}P(\nu)$ is proportional to a linear dimension, implying that $n-1=1$, *i.e.*, $n=2$.

A survey of the solution of the first kind of order $n=1$ and $n=2$ shows that B, BA and $B\mu$ are the functions appropriate to the three boundary conditions respectively. It is easy to verify that these functions do in fact satisfy equation (26) for $j=h^2/(h^2+k^2)$, $j=1$, and $j=(k^2+4h^2)/(k^2+h^2)$, respectively. Taking the three cases in turn, the analysis can now be completed by the straightforward determination of the constant C.

(*i*) $\quad n=1$, $H(\nu)=B_1$ and so $P(\nu)=1$ and $\Phi=CrBB_1\int_\mu^\infty\frac{dt}{B_3{}^3A_3}$

Hence, by (33)

$$\lim_{\mu\to k}\frac{\partial\Phi}{\partial z}=CkA_2\int_k^\infty\frac{d}{dt}\left(\frac{1}{tA_3}\right)\frac{dt}{B_3}$$

Evaluating the integral

$$\int_k^\infty\frac{d}{dt}\left(\frac{1}{tA_3}\right)\frac{dt}{B_3}$$

we obtain

$$\lim_{\mu\to k}\frac{\partial\Phi}{\partial z}=-\frac{C}{A_2}E(h/k)$$

where $E(u)$ is the complete elliptic integral of the second kind. Hence, by (9)

$$V_\infty=-\frac{C}{A_2}E(h/k)$$

or

$$C=-\frac{A_2V_\infty}{E(h/k)}$$

Also, by (29) the potential at the aerofoil equals

$$\Phi=\frac{CrB_1}{kA_2}=\frac{V_\infty}{kE(h/k)}\cdot rB_1$$

which is, in terms of x and y, taking into acount (13)

$$\Phi=-\frac{V_\infty}{kE(h/k)}(A_2{}^2x^2/\beta^2-k^2y^2)^{\frac{1}{2}}$$

Now $\beta=\cot\bar{\mu}$, $k=\cot\gamma$, $h=(\cot^2\gamma-\cot^2\mu)^{\frac{1}{2}}$ and so finally, at the aerofoil

$$\Phi=-\frac{V_\infty}{E'(\cot\bar{\mu}\tan\gamma)}(x^2\tan^2\gamma-y^2)^{\frac{1}{2}}\quad .\quad .\quad .\quad .\quad (34)$$

where the sign of the square root is positive at the top surface and negative at the bottom surface, and $E'(u)$ is the elliptic integral

$$E'(u) = \int_0^{\pi/2} [1 - (1 - u^2)\sin^2\phi]^{\frac{1}{2}} d\phi$$

The pressure difference between top and bottom surface is then given by

$$\bar{\Delta p} = -\frac{2\rho V^2 \alpha \tan^2\gamma}{E'(\cot\mu \tan\gamma)} \left(\frac{r}{(x^2\tan^2\gamma - y^2)^{\frac{1}{2}}}\right) \quad . \quad . \quad . \quad (35)$$

The spanwise lift distribution $l(y)$ is obtained by integrating the pressure difference $\bar{\Delta p}$ along the chords of the aerofoil.

$$l(y) = \int \bar{\Delta p} dx = -2\Phi(c, y) = -\frac{2\rho V^2 \alpha}{E'(\cot\mu \tan\gamma)} \sqrt{(c^2\tan^2\gamma - y^2)}$$

since Φ vanishes at the leading edge. Integrating $l(y)$ across the span, from $y = -c\tan\gamma$ to $y = c\tan\gamma$, we then obtain the total lift

$$L = -\frac{\pi\rho V^2 \alpha c^2 \tan^2\gamma}{E'(\cot\mu \tan\gamma)} \quad . \quad . \quad . \quad . \quad . \quad . \quad (36)$$

The lift curve slope is therefore given by

$$\partial C_L/\partial\alpha = -\frac{2\pi \tan\gamma}{E'(\cot\mu \tan\gamma)} \quad . \quad . \quad . \quad . \quad . \quad (37)$$

in agreement with (1, *i*) above.

Integrating (35) across the span for a given chordwise position, we find that the total lift per unit length of the chord is proportional to the distance from the apex. It follows readily that the chordwise position of the centre of pressure is at two-thirds of the maximum chord, or four-thirds of the mean chord, aft of the apex.

The pressure distribution over a Delta wing at incidence in the case here considered ($\gamma < \mu$) was first determined by the present author[5] by a method essentially equivalent to, but in detail different from, the method used in the present report (see ref. 3). The same result was obtained independently by Stewart[6]

According to (35) the pressure becomes infinite at the leading edge. As in subsonic flow, this entails that the resultant force is no longer normal to the aerofoil and a special "suction-force" has to be taken into account in assessing the drag[3, 5]

(*ii*) We now turn to the determination of the damping derivative in roll. In that case, we take

$$n = 2,\ H(v) = B_1 A_1,\ \text{so that } P(v) = A_1 \text{ and } \Phi = Cr^2 B_1 A_1 BA \int_\mu^\infty \frac{dt}{A_3^3 B_3^3}$$

so that at the aerofoil

$$\operatorname*{Lim}_{\mu\to k} \frac{\partial\Phi}{\partial z} = Crk A_2^2 A_1 \int_\kappa^\infty \frac{d}{dt}\left(\frac{1}{tA_3^3}\right)\frac{dt}{B_3}$$

The elliptic integral on the right hand side equals

$$\int_\kappa^\infty \frac{d}{dt}\left(\frac{1}{tA_3^3}\frac{dt}{B_3}\right) = -\frac{K}{h^2 A_2^4}\left[\left\{1 + \left(\frac{h}{k}\right)^2\right\} E\left(\frac{h}{k}\right) - \left\{1 - \left(\frac{h}{k}\right)^2\right\} K\left(\frac{h}{k}\right)\right]$$

where $K(u)$ is the complete elliptic integral of the first kind.

At the aerofoil, $rA_1 = hy$ so that (10) is satisfied, provided C is chosen so that

$$p = -\frac{Ck^2}{hA_2^2}\left[\left\{1 + \left(\frac{h}{k}\right)^2\right\} E\left(\frac{h}{k}\right) - \left\{1 - \left(\frac{h}{k}\right)^2\right\} K\left(\frac{h}{k}\right)\right]$$

and therefore

$$C = -\frac{phA_2^2}{k^2\left[\{1+(h/k)^2\}E\left(\frac{h}{k}\right) - \{1-(h/k)^2\}K\left(\frac{h}{k}\right)\right]} \quad . \quad . \quad (38)$$

Again, by (29) the value of the potential at the aerofoil is

$$\Phi = \frac{Cr^2B_1A_1}{kA_2^2} = \frac{Chy}{kA_2^2}\left(\frac{A_2^2x^2}{\beta^2} - k^2y^2\right)^{\frac{1}{2}}$$

Substituting the value of C from (38) and recalling that

$$\beta = \cot\bar{\mu},\ k = \cot\gamma,\ h = (\cot^2\gamma - \cot^2\bar{\mu})^{\frac{1}{2}}$$

we finally obtain

$$\Phi = -p\frac{[1-(\cot\bar{\mu}\tan\gamma)^2]\,y\,(x^2\tan^2\gamma - y^2)^{\frac{1}{2}}}{[2-(\cot\bar{\mu}\tan\gamma)^2]E'(\cot\bar{\mu}\tan\gamma) - (\cot\bar{\mu}\tan\gamma)^2K'(\cot\bar{\mu}\tan\gamma)} \quad (39)$$

where $K'(u)$ is the elliptic integral

$$K'(u) = \int_0^{\pi/2}\frac{d\Phi}{[1-(1-u^2)\sin^2\phi]^{\frac{1}{2}}}$$

Similarly as in (*i*), (39) yields the pressure distribution, the spanwise lift distribution, the resulting aerodynamic rolling moment, and finally the damping derivative l_t given in formula (2,*i*), above.

(*iii*) Lastly, we turn to the calculation of the aerodynamic forces acting on a Delta wing in a steady pitch. In that case, we take $n=2$, $H(v)=B_1v$ and so $P(v)=v$ and

$$\Phi = Cr^2\mu vBB_1\int_{\mu}^{\infty}\frac{dt}{t^2B_3^3A_3}$$

Then, by (33)

$$\lim_{\mu\to k}\frac{\partial\Phi}{\partial z} = Crk^2A_2v\int_k^{\infty}\frac{d}{dt}\left(\frac{1}{t^3A_3}\right)\frac{dt}{B_3}$$

The value of the elliptic integral on the right hand side is

$$\int_{\mu}^{\infty}\frac{d}{dt}\left(\frac{1}{t^3A_3}\right)\frac{dt}{B_3} = -\frac{1}{kh^2A_2^2}[\{2(h/k)^2-1\}E(h/k) + \{1-(h/k)^2\}K(h/k)]$$

Also, at the aerofoil, $rv=hx/\beta$, so that Φ satisfies (11) provided C is determined from

$$q = -C\frac{hk^2A_2}{\beta kh^2A_2^2}[\{2(h/k)^2-1\}E(h/k) + \{1-(h/k)^2\}K(h/k)]$$

Thus

$$C = -q\frac{\beta hA_2}{k[\{2(h/k)^2-1\}E(h/k) + \{1-(h/k)^2\}K(h/k)]}$$

so that, by (29), the potential at the aerofoil equals

$$\Phi = Cr^2\frac{B_1}{k^2A_2} = C\frac{\beta hx}{k^2A_2}\{A_2^2x^2/\beta^2 - k^2y^2\}^{\frac{1}{2}}$$

$$= -q\frac{\beta h^2x\{A_2^2x^2/\beta^2 - k^2y^2\}^{\frac{1}{2}}}{k^3[\{2(h/k)-1\}E(h/k) + \{1+(h/k)^2\}K(h/k)]}$$

Substituting $\cot\bar{\mu}$, $\cot\gamma$ and $\{\cot^2\gamma-\cot^2\bar{\mu}\}^{\frac{1}{2}}$ for β, k and h respectively, we finally obtain

$$\Phi=-q\frac{\{1-(\cot\bar{\mu}\tan\gamma)^2\}^{\frac{1}{2}}x\,(x^2\tan^2\gamma-y^2)^{\frac{1}{2}}}{\{1-2(\cot\bar{\mu}\tan\gamma)^2\}E'(\cot\bar{\mu}\tan\gamma)+(\cot\bar{\mu}\tan\gamma)^2K(\cot\bar{\mu}\tan\gamma)} \quad (40)$$

As before, we can now use (40) in order to derive in turn the longitudinal induced velocity, the pressure distribution, the total lift and pitching moment, the chordwise position of the centre of pressure, and finally m_q and $\partial C_L/\frac{\partial\bar{c}q}{V}$ as given in formulæ (3, *i*) and (4, *i*) respectively.

2.3. The definitely supersonic case.

As the free stream velocity V increases, the Mach angle $\bar{\mu}$ of the flow becomes progressively smaller, so that, for a given aerofoil, at sufficiently high speed $\gamma\geqslant\bar{\mu}$ (case (*iii*) of para. 1.3). Under these conditions, the analysis of para. 2.2 above no longer applies, but another method becomes available in its place.

In supersonic flow, any small disturbance can be propagated in the fluid only downstream, and inside the Mach cone issuing from the origin of the disturbance. Also, it is impossible for any disturbance to be propagated across a solid boundary, such as an aerofoil. As a result, the pressure distribution on each of the two surfaces of a Delta wing whose apex semi angle is greater than the given Mach angle, can depend only on the shape of the respective surface, *i.e.*, the pressure distribution on the upper surface is independent of the shape of the lower surface, and *vice-versa*. This "principle of independence" has been applied in ref. 7—following a suggestion by H. B. Squire—to the calculation of the pressure distribution over a flat Delta wing at incidence (under the assumption that $\gamma\geqslant\bar{\mu}$), and the same procedure has been adopted independently by A. E. Puckett.[8] However, the pressure distribution for this case has been calculated earlier by G. N. Ward, using a different method.[9]

In view of the "principle of independence" stated above, we replace the boundary conditions (9), (10) and (11), respectively, by

$$\alpha=\frac{1}{V}\left(\frac{\partial\Phi}{\partial z}\right)_{z=+0}=-\frac{1}{V}\left(\frac{\partial\Phi}{\partial z}\right)_{z=-0} \quad . \quad . \quad . \quad . \quad (9^1)$$

$$py=\left(\frac{\partial\Phi}{\partial z}\right)_{z=+0}=-\left(\frac{\partial\Phi}{\partial z}\right)_{z=-0} \quad . \quad . \quad . \quad . \quad (10^1)$$

and

$$qx=\left(\frac{\partial\Phi}{\partial z}\right)_{z=+0}=-\left(\frac{\partial\Phi}{\partial z}\right)_{z=-0} \quad . \quad . \quad . \quad . \quad (11^1)$$

The boundary conditions at the bottom surface are the same as before, so that the pressure distribution over the bottom surface remains unaffected. The boundary conditions at the top surface are still the same in magnitude, but opposite in sign. As a result the conditions are now symmetrical with respect to the $x-y$ plane.

We proceed to construct the corresponding velocity potentials.

A particular solution of (5) is

$$\left.\begin{aligned}&\Phi=\frac{\sigma}{[(x-x_o)^2-\beta^2\{(y-y_o)^2+(z-z_o)^2\}]^{\frac{1}{2}}},\\&\text{if}\\&x_o>x_1\ (x-x_o)^2-\beta^2[(y-y_o)^2+(z-z_o)^2]>0\\&\Phi=0\ \text{elsewhere}\end{aligned}\right\} \quad . \quad (41)$$

where the point x_o, y_o, z_o, is fixed, and the point x, y, z, is variable. In this case, Φ is said to be the velocity potential produced at a point x, y, z by a supersonic point source of strength σ at the point x_o, y_o, z_o. Given a continuous distribution of sources over the x - y plane or over a portion of it, the resultant velocity potential will be continuous across the x - y plane, while the normal derivative $\partial\Phi/\partial z$ is in general

discontinuous across that portion of the x - y plane on which the sources are situated. Let δ be the value of the jump on passing through the x - y plane,

$$\delta = \left(\frac{\partial \Phi}{\partial z}\right)_{z=+0} - \left(\frac{\partial \Phi}{\partial z}\right)_{z=-0}$$

The relation between δ and the local surface source density σ is $\delta = 2\pi\sigma$ in supersonic flow, as opposed to $\delta = 4\pi\sigma$ under subsonic conditions. Also, the velocity potential due to a distribution of sources in the x - y plane is necessarily symmetrical with respect to the x - y plane so that $\partial\Phi/\partial z$ is antisymmetrical with regard to the same plane. Hence $\delta = 2\left(\frac{\partial \Phi}{\partial z}\right)_{z=+0}$, and so

$$\sigma(x_o, y_o) = \frac{1}{\pi}\left(\frac{\partial \Phi}{\partial z}\right)_{z_o=+0} \quad . \quad . \quad . \quad . \quad . \quad (42)$$

In the cases here considered $\left(\frac{\partial \Phi}{\partial z}\right)_{z_o=+0}$ is given by one of the equations (9^1) - (11^1) at the (normal projection of the) aerofoil (in the x - y plane), no sources being situated elsewhere.

The corresponding velocity potential is

$$\Phi(x, y, z) = \int\int \frac{\sigma(x_o, y_o)\, dx_o dy_o}{[(x - x_o)^2 - \beta^2\{(y - y_o)^2 + z^2\}]^{\frac{1}{2}}} \quad . \quad . \quad (43)$$

Where the integration extends over those points of the aerofoil for which the integrand is real and $x_o > x$. The domain of integration will in general be bounded by a hyperbola and by segments of the leading edges of aerofoil.

To evaluate (43), it is convenient to put

$$x_o - x = \xi,\ \beta(y_o - y) = \eta,\ \beta z = \zeta$$

Then

$$\Phi(x, y, z) = \frac{1}{\beta}\int\int \frac{\sigma(x + \xi,\ y + \eta/\beta)\, d\xi d\eta}{(\xi^2 - \eta^2 - \zeta^2)^{\frac{1}{2}}}$$

For the region in which the integral is to be evaluated, $\xi > 0$. We may therefore put

$$\xi = r\cosh\psi,\ \eta = r\sinh\psi,\ |\zeta| < \tau < \infty,\ -\infty < \psi < \infty \quad . \quad . \quad (44)$$

so that

$$r^2 = \xi^2 - \eta^2 \text{ and } \tanh\psi = \eta/\xi$$

Also

$$\frac{\partial \xi}{\partial r} = \cosh\psi,\ \frac{\partial \xi}{\partial \psi} = r\sinh\psi$$

$$\frac{\partial \eta}{\partial r} = \sinh\psi,\ \frac{\partial \eta}{\partial \psi} = r\cosh\psi$$

so that

$$\frac{D(\xi, \eta)}{D(r, \psi)} = r$$

and Φ becomes, in terms of r and ψ,

$$\Phi(x, y, z) = \frac{1}{\beta}\int\int \frac{\sigma(x + r\cosh\psi,\ y + r\sinh\psi/\beta)\, r dr d\psi}{(\tau^2 - \zeta^2)^{\frac{1}{2}}} \quad . \quad . \quad (45)$$

In particular, as $\zeta \to 0$, we have

$$\Phi(x, y, z) = \frac{1}{\beta}\int\int_S \sigma(x + r\cosh\psi,\ y + r\sinh\psi/\beta)\, dr d\psi \quad . \quad . \quad (46)$$

where the domain of integration S is now bounded by the straight lines $\xi \pm \eta = 0$, and by segments of the leading edge of the aerofoil.

We are now going to carry out the integration for the three problems in turn.

(*i*) For the Delta wing at a small constant incidence we have, by (9^1) and (42), $\sigma = V\alpha/\pi$ and so

$$\Phi(x, y, o) = \frac{V\alpha}{\beta\pi}\int\int_S dr d\psi = \frac{V\alpha}{\beta\pi}\int_C r d\psi$$

where the integral on the right hand side is to be taken round the boundary C of S. The result is

$$\Phi = \frac{V\alpha\,(x + |\,y\,|\,\cot\gamma)}{(\cot^2\bar{\mu} - \cot^2\gamma)^{\frac{1}{2}}} \quad . \quad . \quad . \quad . \quad . \quad (47,\ i)$$

when $|\,x\,|\cot\gamma \leqslant |y\,|\cot\bar{\mu}$, *i.e.* outside the Mach cone of the apex (see Fig. 7), and

$$\Phi = \frac{V\alpha}{\pi\,(\cot^2\bar{\mu} - \cot^2\gamma)^{\frac{1}{2}}}\left[(x - y\cot\gamma)\tan^{-1}\left\{\frac{\cot\bar{\mu} - \cot\gamma}{\cot\bar{\mu} + \cot\gamma}\cdot\frac{x + y\cot\gamma}{x - y\cot\gamma}\right\}^{\frac{1}{2}}\right.$$
$$\left. + (x + y\cot\gamma)\tan^{-1}\left\{\frac{\cot\bar{\mu} - \cot\gamma}{\cot\bar{\mu} + \cot\gamma}\cdot\frac{x - y\cot\gamma}{x + y\cot\gamma}\right\}^{\frac{1}{2}}\right] \quad . \quad . \quad (47,\ ii)$$

when $|\,x\,|\cot\gamma \geqslant |\,y \to \cot\bar{\mu}$, *i.e.* inside the Mach cone of the apex.

The procedure in the remaining cases is quite similar. Thus, for roll,

$$\Phi = \frac{p\,(x + y\cot\gamma)}{2\,(\cot^2\bar{\mu} - \cot^2\gamma)^{3/2}}[x\cot\gamma + y\,(2\cot^2\bar{\mu} - \cot^2\gamma)] \quad . \quad . \quad (48,\ i)$$

when $|\,x\,|\cot\gamma \leqslant y\cot\bar{\mu}$, on starboard ($y > 0$). The potential for the corresponding region on port is of equal magnitude but of opposite sign.

When

$$|\,x\,|\cot\gamma \geqslant |\,y\,|\cot\bar{\mu}$$

$$\Phi = \frac{p\cot\gamma}{\pi\,(\cot^2\bar{\mu} - \cot^2\gamma)^{3/2}}\left\{-y\,(\cot^2\bar{\mu} - \cot^2\gamma)^{\frac{1}{2}}\,(x^2 - y^2\cot^2\gamma)^{\frac{1}{2}}\right.$$
$$+ [x^2 - 2xy\cot\gamma + y^2\,(2\cot^2\bar{\mu} - \cot^2\gamma)]\tan^{-1}\left[\frac{\cot\bar{\mu} - \cot\gamma}{\cot\bar{\mu} + \cot\gamma}\cdot\frac{x + y\cot\gamma}{x - y\cot\gamma}\right]^{\frac{1}{2}}$$
$$\left. + [x^2 + 2xy\cot\gamma + y^2\,(2\cot^2\bar{\mu} - \cot^2\gamma)]\tan^{-1}\left[\frac{\cot\bar{\mu} - \cot\gamma}{\cot\bar{\mu} + \cot\gamma}\cdot\frac{x - y\cot\gamma}{x + y\cot\gamma}\right]^{\frac{1}{2}}\right\}$$
$$. \quad (48,\ ii)$$

Finally, for pitch, we obtain

$$\Phi = \frac{q\,(x + |\,y\,|\cot\gamma)}{2\,(\cot^2\bar{\mu} - \cot^2\gamma)^{3/2}}[y\cot\gamma\cot^2\bar{\mu} - x\,(\cot^2\bar{\mu} - 2\cot^2\gamma)] \quad . \quad (49,\ i)$$

when $|\,x\,|\cot\gamma \leqslant |\,y\,|\cot\mu$, and

$$\Phi = \frac{q}{\pi\,(\cot^2\bar{\mu} - \cot^2\gamma)^{3/2}}\left\{x\cot\gamma\,(\cot^2\bar{\mu} - \cot^2\gamma)^{\frac{1}{2}}\,(x^2 - y^2\cot^2\gamma)^{\frac{1}{2}}\right.$$
$$+ (x + y\cot\gamma)[y\cot\gamma\cot^2\bar{\mu} - x\,(\cot^2\bar{\mu} - 2\cot^2\gamma)]\tan^{-1}\left[\frac{\cot\bar{\mu} - \cot\gamma}{\cot\bar{\mu} + \cot\gamma}\cdot\frac{x - y\cot\gamma}{x + y\cot\gamma}\right]^{\frac{1}{2}}$$
$$\left. + (x - y\cot\gamma)[-y\cot\gamma\cot^2\bar{\mu} - x\,(\cot^2\bar{\mu} - 2\cot^2\gamma)]\tan^{-1}\left[\frac{\cot\bar{\mu} - \cot\gamma}{\cot\bar{\mu} + \cot\gamma}\cdot\frac{x + y\cot\gamma}{x - y\cot\gamma}\right]^{\frac{1}{2}}\right\}$$
$$. \quad (49,\ ii)$$

when $|\,x\,|\cot\gamma \geqslant |\,y\,|\cot\mu$.

It will be observed that, as in the quasi-subsonic case, Φ again vanishes everywhere at the leading edge of the wing.

In the modified problem, the value of Φ as given above is obtained when approaching the aerofoil from above or from below, but in the original problem,

$$\Phi(x,\ y,\ -0) = -\Phi(x, y,\ +0)$$

and so, taking (8) into account, the pressure difference between the top and bottom surfaces is given by

$$\bar{\Delta p} = -2\rho V\left(\frac{\partial\Phi}{\partial x}\right)_{z=0}$$

where Φ is given by one of the equations (47, *i*) - (49, *ii*). The other data can now be determined exactly as in the quasi-subsonic case.

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AIRCRAFT SERVICING

by

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INTRODUCTION.

THE method of achieving efficient and economical servicing with the maximum rate of utilisation of aircraft, is a problem which always confronts the aircraft operator, either a private owner flying his modest machine or a national airline with a variety of large and expensive aircraft. From the point of view of safety, their problems are the same.

Efficient servicing is closely related to safety in so far as the airworthiness aspect is concerned and, under present servicing methods, the direct cost of providing this aspect of safety may well amount to some 20 per cent. of operating costs; to this must be added the delays inseparable from major maintenance work which can reduce earning capacity as a result of the time taken to rectify defects. Invariably the cost to the private owner is higher than he had expected and, unless his aircraft is maintained by a competent servicing organisation, it can be subjected to frequent bouts of unserviceability—usually when he needs it most.

Flying clubs and charter companies have the same problem varying in complexity and scope in relation to their operating conditions and numbers and types of aircraft operated.

In studying this problem, it is convenient at this stage to review the development of servicing methods.

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THE DEVELOPMENT OF SERVICING.

Until comparatively recently comprehensive maintenance schedules were an exception rather than the rule and the servicing of aircraft implied an inspection procedure, the primary consideration of which was to determine by visual examination which parts had become unserviceable. Coupled with any adjustments needed were defects reported by members of the operating crew. In the main, the keynote was "let us look to see if anything has gone wrong or is going wrong so that we may put it right." Daily and periodic inspections were made by engineers, largely in accordance with the degree of basic knowledge they possessed supported by brief instructions issued by the manufacturer.

This method of inspection was haphazard in that an unnecessary amount of time was spent in inspecting parts which did not require inspection, while parts which required attention did not receive it. In consequence, no accurate indication could be given at the beginning of the servicing of the length of time which might be spent rectifying defects *yet to be discovered* and which might call for spare parts not held in stock, or involve widespread dismantling of aircraft urgently needed for service.

The application of maintenance schedules to aircraft operated by the civil elementary flying training schools, the issue by manufacturers of servicing instructions of wider scope and the methods introduced by the principal airline companies, whereby the annual certificate of airworthiness overhaul was done by instalments throughout the year, in addition to daily routine checks, all had a

marked effect and produced a "servicing conscience" which kept in the foreground the need for better and easier maintenance.

Despite these improving factors many operators were content, on daily and minor routine checks, to correct only those points which were essential to the comparatively safe flight of the aircraft, and an annual overhaul before the renewal of a certificate of airworthiness became a recognised practice. Thus, there was a tendency to leave to this annual overhaul jobs which might not be convenient at other times. Because of this and the generally unsatisfactory method of recording the work done during the previous year, the work involved was extensive, requiring considerable expenditure of labour, spares and, above all, time, with, consequently, loss of revenue. In addition, because there was no evidence of systematic and scientifically devised inspection or rectification operations, the aircraft surveyors of the British Corporation Register, Lloyd's Register and later, of the Air Registration Board, were forced to call for extensive dismantling of the structure and components in order that the condition of the aircraft could be accurately determined. Such a degree of dismantling was necessary only because *no one knew or could prove* what was the real condition of the aircraft. In addition to being unnecessary, much of this dismantling did little good to the general structure.

The length of time an aircraft was off service for the overhaul depended to a great extent on the knowledge which the engineers responsible for the servicing and overhaul of the aircraft had acquired as a result of their practical experience with the type of aircraft concerned. It cannot be denied that, generally speaking, the manufacturers provided little information in the way of servicing and instruction manuals to guide the engineers in their work.

During the war the Royal Air Force introduced various systems for accurately planning the maintenance of aircraft. The various stages in a complete inspection operation, whether daily, minor or major, were examined and analysed both from the "time spent" aspect and the location of the jobs on the aircraft. The two main considerations were, first to do each individual inspection operation in the shortest possible time consistent with a high degree of efficiency, and second, to arrange the route taken by tradesmen in the various branches who must visit the same location on the aircraft (*e.g* cabin) so that each followed the other at precise intervals of time. Every effort was made to avoid the possibility of two men being in the same place at the same time to perform different operations. In addition, the concentration of tradesmen was such that their work remained within a limited area. This method resulted in a quicker turn-round of aircraft with a saving in manpower, and on this score alone could be justified as being an important advance in technique.

Despite this advance the basic principle remains the same. There is still the search for what has gone wrong followed by rectification of the defects disclosed; there is still the same unpredictable amount of time expended in putting right defects unpredictable in number, in consumption of spares and in the delaying effect on planned operational flying.

The quality of maintenance varies directly with the skill and knowledge of the staff employed and although, with highly skilled mechanics and inspectors, the present system can provide safety, it is only at the expense of time. Maintenance staff lacking skill and knowledge are prone to reject a high proportion of the parts examined in position or removed for inspection because their ability to assess safe life is not developed to the high degree essential for the economic application of this system. Their attitude becomes, and rightly so, "when in doubt change the part," a method which can become expensive in practice and is economically unsound.

It is unreasonable to expect all maintenance personnel to possess vast experience and precise judgment, but it is time that a large measure of this fallible human element was removed by the application of methods more in keeping with the present state of scientific and industrial development.

A NEW APPROACH.

The Air Registration Board was aware of the inefficiency of the pre-war methods, but because of the depletion of staff during the war years, was not able to do much about

new methods. Shortly after the end of the war it introduced the method of servicing aircraft in relation to approved maintenance schedules, but from that time until quite recently, maintenance schedules took account only of routine servicing and not of the work which in the past had been done during the annual overhaul.

It was apparent that there was a need to relate the overhaul of aircraft to "flying hours" rather than "calendar time" and the Air Registration Board has extended the scope of maintenance schedules. At the present time a number of operators are overhauling their aircraft in relation to flying hours, so that in many cases the "annual overhaul" has disappeared and aircraft are only off service for a few days when the certificate of airworthiness is due for renewal, instead of for weeks or, in some cases, months under the old method.

Certificates of safety for flight are related at present to calendar time, irrespective of the number of hours flown by the aircraft concerned, but the time has come when they should be related to the flying hours. Maintenance schedules should provide for work to be done to ensure that the aircraft is safe for flight for a given period and for work to be done to avoid the need of an annual overhaul.

Ideally, the requirements of these two cases should be co-ordinated and on occasions when the check before the issue of the certificate of safety for flight is made, additional work of a replacement nature should take place. The feasibility of this method may be developed from systems already in operation.

In one such system an addition to the maintenance schedule provides for the overhaul to be done in four instalments, within a specified period of flying hours irrespective of calendar time, plus a general examination by an A.R.B. Surveyor before the recommendation for the renewal of the certificate of airworthiness. Under this system, lengthy periods may be spent on the ground and the total lapsed time spent in making these overhauls by instalments may well be not far short of the time which would have been spent had the overhaul been completed in one stage.

Obviously, the greater the number of instalments the shorter will be the time taken by each. With the present utilisation rate of aircraft (from four to ten hours daily) there seems to be ample time for the high-frequency short-period servicing scheme. If the worst possible case is taken where the overhaul would take three months, this represents an expenditure of only six hours in each 24-hour day throughout the year. Assuming a utilisation rate of 16 hours a day is reached (and this may be a decade ahead), there still remains a period of eight hours a day, which represents one-third of the year or a potential period of four months on the ground during which replacements and checks can be made.

This new approach, however, can only be described as an interim method of servicing because, apart from engines, variable-pitch propellers, and certain instruments which have a definite prescribed life between overhaul, the life of the remainder of the aircraft is related to a common period of *X* hours, which clearly indicates that a number of components are overhauled before it is really necessary.

On somewhat similar lines, but less rigorous in its provisions, is the servicing scheme for private owners only. This applies to aircraft not exceeding 3,500 lb. all-up weight. The scheme offers advantages to those private owners who may not do much flying each year and who feel that they would be satisfied with a more modest servicing scheme.

A NEW CONCEPTION.

An entirely new conception of servicing methods is essential, particularly if expensive high performance aircraft are to be operated economically at a high rate of utilisation. No longer should it be necessary to remove parts after completing relatively short periods in service in order to determine whether they are still serviceable; and no longer should it be necessary to "jack-up" an aircraft at regular periods in order to determine whether there is wear in pivotal points or float in bearings of landing gears. It must be known with a reasonable degree of accuracy before an aircraft is put into service for the first time how long the numerous components, instruments, accessories and so on, will remain serviceable in normal operating

conditions before they need overhauling or replacing.

There are several factors which will affect the life in service of parts in relation to conditions under which aircraft are operated, but in determining the basic life, operating conditions could be ignored and corrections could be made to results which were obtained by other methods.

There are several methods by which the safe life of a part in service could be determined, such as:—

(*a*) by laboratory research,

(*b*) by intelligent guess-work,

(*c*) by accurate design for maintenance,

(*d*) by operating experience, *i.e.* the consumption of parts against hours flown.

Of these four methods, laboratory research appears to be the most desirable method, but if done by the individual manufacturer it would be confined to a study of those parts peculiar to his own type and his own design. His field would thus be limited to mechanism favoured by his own design staff and would not necessarily be representative of the technique of the Industry as a whole. It would be necessary for the suppliers of accessories (compressors, instruments, hydraulic gear, and so on) to provide with each item estimates of the life of accessories, together with lives of individual items embodied in each accessory and full servicing instructions regarding the diagnosis and rectification of all possible defects.

Considering the Industry as a whole, much over-lapping and waste of effort would result unless a means of co-ordinating the results of such research were found. It is suggested that there is a case for the establishment of a central research laboratory sponsored and financed by the Society of British Aircraft Constructors, to which the results of all research or statistical information might be sent for analysis to augment the laboratory's own discoveries, and from which information thus culled from a diversity of sources might be disseminated.

The amount of work which this establishment would have to do is not so formidable as would at first appear, if it is agreed that it would be the responsibility of the manufacturers of the accessories to provide the necessary data in relation to their own products.

Any objection to the proposal to establish this central research laboratory may well be on the ground of cost, but considering the cost of servicing aircraft under past and present methods, there is little doubt that the ultimate result would mean a substantial reduction in the cost of servicing.

Intelligent guess-work can be based only on wide experience in the servicing of aircraft mechanisms and components. Invariably such estimates are affected by personal prejudices and would relate only to the types of aircraft within the scope of an individual's own experience. It would not be possible in this way to provide correction factors for varying climatic or operating conditions or for peculiarly concentrated forms of operation, such as frequent short hauls or flying training. Practical experience could decide for instance, that a certain pivotal bolt would last, say, at least 500 hours, but would hesitate to extend this life to 1,000 hours without proper investigation.

Associated with the replacement of parts which have completed their overhaul life, is the time taken to effect their replacement. One often hears the phrase "design for maintenance," but the new conception should be "design for little or no maintenance" or, to put it another way, "design so as to provide for the maximum possible life between overhaul." With this should be coupled "design so that a part can be replaced in the minimum time."

It is not suggested that this is easy to achieve; it is but one of the designer's considerations and may conflict with other requirements. Also, a more economic use of material and a wider design choice is possible if the design is not influenced by maintenance requirements. Points of major importance such as accessibility, ease of removal of components and assemblies, grouping of "topping-up" points (de-icer fluid, hydraulic fluid, compressed air, and so on), are now receiving somewhat belated attention, but design to provide specific lives for all components subject to wear is in its infancy. It should be possible to design pivotal bolts or pins, their bearings and the items borne therein, to have specific safe lives on which reliance similar to that achieved in aero-engine practice can be placed. This may have

been done, but if it has, the information has not been supplied to those responsible for servicing.

In designing for maximum life between overhauls, regard must be paid to built-in assemblies which may have a relatively short life and thus may shorten the life of the assembly as a whole. A mainplane with integral tanks is a fair example since the overhaul life of such a mainplane would be restricted to the life of the integral tanks. Another example is a power unit or power egg, the various components of which at the present time have a different overhaul life. From the operator's point of view it would be most convenient, and certainly more economical, to replace the unit as a whole rather than replace the engine at *X* hours, the propeller at *Y* hours, the radiators at *Z* hours, and so on.

Other points which must be considered are the grouping of accessories or systems with a common overhaul life so that one group of accessories or one system or part of a system can be replaced without disturbing another group of accessories or another system.

The final method, as regards the actual consumption of spares per aircraft in service, is less formidable than at first appears. In any well organised base the issue of replacements would be recorded against the aircraft concerned during normal stores procedure. Because hours flown by an aircraft are also recorded, an examination of the documents would show up the life of a part in hours flown and in lapsed time.

The complexity of such investigation would vary with the type of aircraft and the number in service. The clerical work would be relatively heavy in the early stages of the investigation, but as more knowledge became available and lives of components were established, there would be a gradual reduction in staff until the ideal case of "knowing it all" might be reached, well within the normal accepted life of an aeroplane.

The magnitude of normal servicing work is accepted without question and yet the proposed use of say, at the most, the equivalent of one-tenth of the existing labour force on statistical work would probably be viewed with alarm. A reasonable assumption is that the cost of this work would be directly related to the amount of maintenance work done, and the investigation viewed purely as a means to an end. The accrued benefits, however, would exceed the statistical expense in most cases.

Operators who have had some experience with a particular type should be in a position to introduce this method, but due regard should be given to the new types of aircraft which will be coming into service in the next year or so. Laboratory research on the types by the manufacturer and operating experience of other types, should be combined so that these new types can go into service with the expectation of attaining a high rate of utilisation.

With the high rate of utilisation expected from aircraft of the future it may well be that the maximum time such aircraft spend at their base for servicing will not exceed eight hours at any given time. All work in connection with the replacement of time-expired parts must be completed within this period.

The main servicing hangars would thus house an aircraft for a period not exceeding eight hours, while skilled personnel removed the components which had completed their overhaul life and replaced them with serviced items. No other work, apart from cleaning or topping up, etc., as a general rule, should be necessary, and any instrument, accessory, or item of equipment reported by the crew as being defective should be replaced. The principle should be no work to be done on the aircraft which could better be done in a specialist's workshop.

All components, assemblies, accessories, and so on, on completing their overhaul life should be transferred to the appropriate specialists' workshops, where they should be overhauled ready for replacement.

Since the flow of work could be assessed within fine limits, high efficiency would be achieved in shop loading and technique but such a system would only be possible if an adequate flow of replacements were available, and operators would require as part of their initial expenditure adequate spare components and assemblies. These spares would be needed at some time during the life of the aircraft; therefore this outlay would be fully justified.

The operator with a large fleet of aircraft would have no difficulty in establishing and

operating the necessary specialist workshops, but the case of the operator with one or just a few similar aircraft is much different. He would not feel disposed to lay out capital on a wide range of spare components but would prefer to buy them out of revenue. Neither would he have the premises, the equipment nor the technicians in a range wide enough to cope with all contingencies. Therefore, he would be inclined to use the old methods of maintenance unless the manufacturer or a servicing organisation offered a servicing scheme, including the supply of replacement parts, to synchronise with his utilisation rate and maintenance system.

It is feasible that, for the private owner, those manufacturers who are aware that it is after-sales service which keeps aeroplanes in the air, may offer for sale not only aircraft but also, a system of continuous servicing operations related to flying hours and a guaranteed supply of replacement parts of known cost.

(*Correspondence will be welcomed on this paper.*—EDITOR.)

WHY SHEAR WEBS?

by

H. L. COX, M.A., F.R.Ae.S.*

AT a section of a cantilever distant x from its tip, let the depth of section be h and the bending moment M. Then if all the resistance to bending is concentrated in the flanges the loads in the flanges are $\pm M/h$. If the flanges are inclined at a small angle α these flange loads have a component transverse to the cantilever axis of magnitude $M\alpha/h$. Now $\alpha = dh/dx$ and the shear load at section x is dM/dx. Therefore the shear load on the *web* of the cantilever is $dM/dx - (M/h)\,dh/dx$, and, if h/M is constant, the web is not sheared at all.

For a cantilever loaded by a shear load at its tip, M is proportional to x and we need a straight taper to zero at the tip. This case is familiar, but the more general application of the rule is not. For instance, if the loading on the cantilever is $w\{(a/l)x+b\}$, which represents a uniform pressure w on a wing of length l tapering from width $a+b$ at the root to width b at the tip, the shear load

$$F=\int_0^x w\{(a/l)x+b\}\,dx$$
$$=(wx/2)\{a(x/l)+2b\}$$

and

$$M=\int_0^x F dx=(wx^2/6)\{a(x/l)+3b\}.$$

In this case in order to free the web completely from shear, the taper must be concave and the wing tip becomes very thin; this is illustrated for the case $b/(a+b)=\frac{1}{3}$ below and by Fig. 1.

So violent a taper over the wing tip may be impossible for aerodynamic reasons; but modification by increase of the thickness h towards the tip causes shear loads in the web only outboard of the first section modified. The structural problem of providing resistance to the fairly light shear loads in these shallow webs is quite trivial. Shear webs at the root will still be required to provide against torsion; but the shear stresses in the webs due to torsion will normally be much less than the stresses which would result if the whole shear corresponding to the lift were carried by the web.

Disregarding the torsion case for the moment, a properly tapered cantilever has no need of *shear* webs. On the other hand the webs have another essential function which is not generally recognised; that is to stabilise the compression flange against buckling. This function has been demonstrated by a test on a straight tapered cantilever loaded by a shear load at its tip. This model had flanges of cardboard and two webs of thin paper. The cantilever was about $6\frac{1}{2}$ in. long and roughly square in section, tapering from 2×2 in. at the root to $\frac{1}{4}\times\frac{1}{4}$ in. at the tip. The cardboard flanges had a strength of about 90 lb. per in. width and the paper webs about 7 lb. per in. width.

Under shear load applied at the tip, the webs were not entirely free from shear, because the condition M/h constant was not

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	(x/l)	1	0.9	0.8	0.7	0.6	0.5	0.4	0.3	0.2	0.1
Taper for no shear	(h/h_o)	1	0.778	0.589	0.431	0.302	0.200	0.122	0.065	0.027	0.006
Straight taper of same area of web	(h/h_o)	0.6	0.54	0.48	0.42	0.36	0.30	0.24	0.18	0.12	0.06

exactly fulfilled. The divergence from this condition was greater near the tip than near the root and accordingly the webs were more severely sheared near the tip. Under test, buckling of the webs developed near the tip, but the web near the root remained plane. The cantilever failed by buckling of the compression flange inwards, the web crumpling under the compression thus applied to it. The maximum load carried was 8.1 lb. In the final collapse the webs were slightly torn transversely to the axis of the cantilever and in way of the main buckle in the compression flange; these tears were clearly due to the extension of the paper webs along the cantilever axis as a result of the flexure of the compression flange. Otherwise the webs were undamaged.

The degree of support required by the compression flange to prevent its buckling is fairly considerable; but attachment by a web to the tension flange is an efficient means of stabilisation and the requirement concerns mainly the stiffness of the web to extension or compression between the flanges. If the web itself remains flat, the thickness of web required is quite moderate; for instance in aluminium alloy two webs each 0.25 in. thick should be sufficient* to stabilise a compression flange 60 in. wide by 0.08 in. thick stressed to 40,000 lb./in.^{2} (total load 85 tons). (Incidentally this question of stabilisation involves consideration of wing depth and cover thickness and indicates optima for both these variables according to the bending moment to be carried.) In present designs the webs provided to carry shear and torsion would normally meet this requirement.

At the same time a compression surface 0.08 in. and 60 in. wide is not completely stabilised by two webs spaced (say) 40 in. apart. In fact, to remain unbuckled at 40,000 lb./in.^{2}, the 0.08 in. sheet requires stringers at about 2½ in. spacing; the stringers require ribs to limit their free length, and so a conventional design results.

On the other hand the two webs could theoretically be divided into 24 each being 0.02 in. thick, and the only loss by this modification should be a reduction of torsional stiffness by about 25 per cent. The snag is that, even if shear were completely eliminated, these thin webs would become unstable under the cross thrust between the tension and compression flanges. For instance, if the two flanges are 12 in. apart and if both have the same area of 4.8 in.^{2} and are stressed to 40,000 lb./in.^{2}, the thrust on the webs would set up a stress of 250 lb./in.^{2}, whereas the webs 0.02 in. thick and 12 in. deep would buckle at a stress of about 20 lb./in.^{2}. The condition is actually

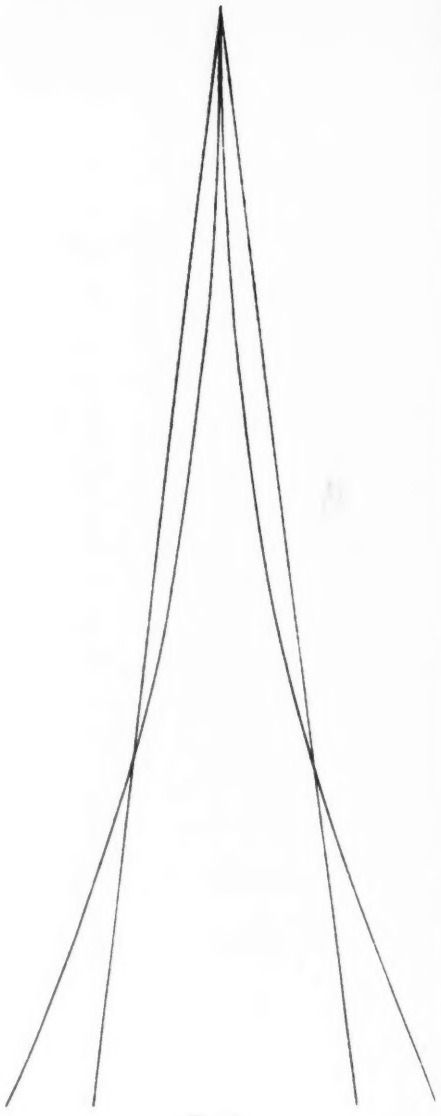

Fig. 1.

* The figure quoted is an upper limit; rather thinner webs might suffice.

rather worse than this, because if the flanges are straight the web is subjected also to shear stress, while, if the wing is tapered progressively to eliminate shear stress, the compression flange is concave and subjects the web to a still higher cross thrust.*

The difficulty can readily be met by curving the axis of the cantilever, so that the compression flange becomes concave to the web. Then under load the two flanges tend to separate and the web, subjected to cross tension, is no longer liable to collapse. This has been demonstrated by a test on a cardboard and paper model exactly similar to that previously described, except that the axis was curved to a radius slightly greater than the length of the beam. Under a shear load applied at the tip this curved cantilever behaved in very much the same manner as the straight one; shear buckles developed in the web near the tip (as a result of inaccurate fulfilment of the condition Mh=constant) but again the web near the root remained flat. Although the compression flange buckled between the webs, there was no sign of overall instability and the failure at a load of 14.6 lb. was by tearing of the webs near the tip along the edges of the compression flange; the computed value of the stress in the web at the point of failure was 8 lb./in., which agrees closely with the measured strength of the paper (7 lb./in.).

An exactly similar curved cantilever with two webs of thicker paper having a strength of 33 lb./in. failed at a tip load of 25 lb. by tearing of the tension flange; the computed stress in this flange was about 80 lb./in., which reduction from the measured value of 90 lb./in. is probably attributable to uneven distribution of load.

On the basis of this theoretical survey and these experimental results it is suggested that multispar wings with very thin webs are feasible, provided that the wing axis may be curved in the plane of the webs; and it is claimed that the total weight of the webs need not exceed one half of the total weight of the wing cover. The curvature of the wing axis required is greater near the root than at the tip, but the least radius need not be less than the (half) span of the wing; suitable forms raise the wing tip above the wing root by about $\frac{1}{4}$ to $\frac{1}{3}$ the half span. The detail of attachment of the wing to the wing root fixing appears to offer advantage rather than disadvantage over conventional design.

Preferably the wing depth should be adjusted to meet the condition Mh= const. for the maximum lift case at least approximately for the wing root, say to 40 per cent. of the span. However, this refinement may not be necessary if the cross tension due to wing curvature may be made great enough more than to annul the cross tension due to development of buckles due to shear. In this respect considerable use can be made of certain considerations affecting the design of the wing root fixing.

* The concavity of the tension flange results in a cross tension in the web, and this tension combined with the thrust from the compression flange sets up a secondary system of shear stress in the web. The actual stress system is rather complex; but the general conclusion that progressive taper results in heavier stresses in the web is justified.

CORRESPONDENCE

EVOLUTION OF AN AEROPLANE—TEST PILOTS

Seldom have I read such a preposterous statement as that contained in your correspondent Mr. Master's letter in the September 1948 JOURNAL that "test pilots were employed solely to take the risks on behalf of the technicians."

Test pilots take risks either because they like doing so or because they find they can make the best living that way—a living much above that of the technicians he appears to despise. Mr. Masters may not be aware that it has even been known for technicians to *lose* their lives because of lack of skill or judgment of test pilots.

Whatever the quality of the output of the E.P.T.S. there was a time when it was the lot of the technicians to convert the raw ex-Flying Officer into some semblance of a critically minded *test* pilot. In my experience this has not always been an easy job, for a fund of aeronautical lore is often mistaken for a precise understanding of applied aerodynamics. A subjective interpretation by the test pilot of the behaviour of an aircraft, instead of an objective statement, can be a considerable hindrance to the satisfactory solution of the many problems confronting the technician. As Mr. P. G. Lucas, at a lecture about three years ago pointed out, test pilots as a class were so deficient in the ability to explain such matters that they invented a jargon of their own—and the technician was expected to acquire an understanding of it, through patient interrogation, if he were to get to the bottom of mysteries of the behaviour of his aircraft.

A. C. Brown, F.R.Ae.S.

AIRCRAFT ACCIDENTS

The saving of life obviously must be tackled from two angles, namely the prevention of acidents and the taking of adequate precautions in case of an accident. The situation is analysed as follows:—

Ground Control

The progress of development of blind landing aids should be constantly under review and encouragement given to the system which demands the least skill and concentration from the ground controller and flight captain.

The senior aerodrome control officer should be empowered to:—

1. Close the aerodrome when local conditions, in his opinion, are too hazardous for safe landings.
2. Order an incoming pilot to an alternative airport where it is known that suitable landing conditions prevail or, if necessary, to Manston where FIDO is installed.

Just as the captain of a ship hands over the responsibility to a pilot at the approaches to a port, so the captain of an aircraft should be subordinate to the senior airport controller on entering the control zone.

Flight Crew.

All crew members should be capable of locating and operating, in darkness, all doors and emergency exits and all life-saving equipment provided on all types of aircraft on which they are employed. Proof of this proficiency should be demonstrated at adequate intervals in a practical manner.

The renewal of pilots' "B" licence should be dependent, among other things, on the completion of regular and successful Link Trainer practice. Pilots should be encouraged to use beam approaches on all possible occasions in order to familiarise themselves with the system and to gain confidence in its use.

Passengers and Safety Equipment.

On all flights on which lifebelts are mandatory, the passengers should put them on, in a deflated condition, before entering the aircraft and the steward should see, as each passenger embarks, that the equipment is properly worn. In an emergency, the discharging of the CO_2 bottles should be supervised by the steward.

Something more indelible than notices alone, should be employed to impress on the minds of passengers the correct use of safety equipment. On shipboard the passengers indulge in boat drill and, although this is impracticable in an aeroplane, there is nothing to prevent a film from being shown, either in flight or before embarkation, indicating the location function and operation of all exits, safety belts, lifebelts, dinghies, hand fire extinguishers and any other safety equipment used on the type of aircraft in which the flight is being made.

I agree that all passenger seats should face aft and, further, so should those of the steward and all members of the crew except, of course, the pilots. All seats should withstand a deceleration of 25G and, besides the standard safety belt, be fitted with a leg strap to keep the legs down, for use when landing or taking off.

The idea is to endeavour to ensure that at least one crew member is available to shepherd the passengers through the exits and prevent panic. To this end, all doors should open outwards. All exits should be tested as part of the daily inspection of the aircraft for fitness of flight.

Crash Fires.

The inflammable fluids carried in aircraft are reasonably safe until a minute spark gets among a mixture of their vapours and air. It would appear that a source of ignition is more dangerous than an inflammable material.

Fires are caused by an inflammable material coming into contact with a source of ignition. In the event of a crash, the material can be provided by:—

1. Burst tanks containing fuel, oil, de-icing fluid or hydraulic fluid.
2. Crushed bag-type tanks causing fluid to be forced out of vent pipes.
3. Ripped-off or wrenched tank fittings.
4. Ruptured or severed pipelines containing inflammable fluids.
5. Damaged and leaking hydraulic components and/or pipelines, containing fluid under high pressure.
6. In conjunction with No. 4, cocks not closed.

1 *and* 2. All tanks containing inflammable fluids should be crash-proof, including the vent system, and should all be fitted with cocks. That is, a cock, in each case, should be fitted direct to each tank.

3. External fittings should be so placed on each tank that they are not vulnerable and should not protrude beyond surface skins. The buckling or collapse of surrounding structure should not tend to shear off or sever external fittings.

4. Pipelines containing inflammable fluids should be so installed that there is a minimum possibility of damage by being wrenched, sheared or severed by surrounding structure. In this connection, pipelines should not be run on the face of spars.

5. Hydraulics should be superseded by pneumatic systems or, alternatively, a non-inflammable fluid should be devised.

The evolution of non-inflammable de-icing fluids should be investigated concurrently with thermal de-icing research.

6. All cocks should be closed automatically at decelerations of the order of 3G. and this should not be effected electrically. If mechanically-operated cocks are employed they should not be capable of being turned on by the effect of wrenching or impact on their controls.

Electrically-operated cocks should be retained in the open position by an energised solenoid, operating against a spring. Mechanically-operated cocks should have their controls divorced from the cock lever while the remote controls should be kept away from the more vulnerable positions and made to move at right angles to the flight path.

The source of ignition can be supplied by:—

1. Electrical leads wrenched or severed with "live" circuits.
2. Hot engines.
3. Hot exhaust systems.
4. Cabin heater systems.
5. The impact of ferrous metal on stone.
6. Faulty electrical components causing short circuits.

Elaborating the above:—

1 *and* 6. Although electrical circuits are fused, it can easily be proved that considerable short-circuiting can exist without the fuse blowing. Fire extinguishing bottles should be operated mechanically in a crash but electrical operation can still be employed for the purpose of intentional suppression of engine fires in flight. In a crash the batteries should be disconnected automatically from all electrical circuits at a forward deceleration of 3G. Generators, too, should disconnect where turbo-jet engines are installed.

2, 3 *and* 4. Apart from the existing arrangements, all systems employing inflammable fluids should have solenoid-operated or other automatically-closing cocks in positions which prevent excessive quantities of fluid being able to flow into a fire zone in the event of ruptured pipelines.

5. As far as is practicable, items of a ferrous nature should be located so that contact with the ground, in a belly landing, is minimised.

To contain all inflammable fluids and, at the same time, render all electrical circuits dead, should result in a greater factor of safety from crash fires.

Miscellaneous Equipment.

A radio altimeter should be a mandatory instrument for flight in Q.B.I. conditions.

A radio transmitter, situated in the least vulnerable position, usually the tail end of the fuselage, should operate at 3g., on the international distress channel and its flexible mounting should withstand decelerations of 25g. in any direction. A "fix" could thus be rapidly obtained and a "pin-point" location given to rescue parties. This radio should be a self-contained unit operating on dry batteries.

J. Smith, A.R.Ae.S.

MAINTENANCE DESIGN QUESTIONS

The correspondence and short papers in the JOURNAL on aircraft maintenance have been of great interest, especially to one who has been employed on the construction, repair and maintenance of aircraft for the past thirty-five years.

During the course of years, as aircraft have become more and more complicated and expensive and are still incapable of showing a profit as a commercial undertaking, it might be thought that a real effort would have been made to reduce the time spent on maintenance, which at present causes an aeroplane to be merely capital lying idle for a large part of its life.

I believe the solution lies in deleting the need for maintenance altogether.

In the great majority of cases, the routine scheduled inspections do not reveal any defects and the continual periodic opening up and dismantling of the various parts and installations of the aircraft do more harm than good. If the scheduled

inspections do reveal defects, then the designer has failed to produce a serviceable aircraft, because adjustments and replacements ought not to be necessary at such short intervals.

It seems extraordinary that it should be necessary to make a complete inspection of the flying controls every 40 hours or so, when their motion is comparatively slow and they have to transmit the forces imposed on them by one man only, whereas the major internal components of an engine, which are in violent motion and carry great stresses, are permitted to operate for periods varying from 400 to 1,200 hours, without any inspection.

It should be possible to design a flying control system which will operate for 1,000 hours without attention and although it would be a little heavier, perhaps, than the layouts at present used, the loss of payload would be more than offset by the greater serviceability.

Recently, I made an inspection on a small modern four-seater aeroplane on which the control operation was by rods with spherical sockets at each end which engaged ball joints on the levers. A certain amount of slack could be felt at the aileron trailing edges which was traced to slight wear at each of the ball and socket joints, which did not appear to have had much in the way of lubrication since being installed. There was no way of taking up the play except by fitting new ball and socket joints throughout and as the aircraft had done less than 200 hours flying, it would appear that the control design could be improved.

On the same aircraft the shock absorber was connected to the wheel member by a pin, the holes housing which were so worn that they were about $1\frac{1}{4}$ times the pin diameter. The bearing area was obviously inadequate to carry the load imposed on it and the strut fittings, having to be replaced complete, were an expensive item.

I have always found that a well-designed flying control system of the old-fashioned cable and pulley type is the most satisfactory for aircraft up to about 9,000 lb. Also, it has the advantage that a little slack in the pins will not affect the operation of the controls as the pull is always in one direction and a drum of the appropriate size of cable is all that is needed for replacements. This type of layout, if designed with easy cable runs and large diameter plastic pulleys with grease-packed ball races, appears to be good for about 1,000 flying hours without replacements. It is not improved by introducing chains and sprockets in lieu of the pulleys.

On aircraft employing the push-and-pull rod system, much trouble is often caused by wear at the various joints. As is well known, ball bearings are not satisfactory when used at points where only a partial rotation takes place. The needle roller bearing is suitable for this class of work and if push-and-pull rod ends and rock shafts and levers, etc., were made with well designed, dust-tight housings for this type of bearing they should operate for many thousands of hours without attention.

The hydraulic flow control valve is a nuisance which has been with us for ten years or more without having been noticeably improved. It is quite likely that most designers do not know what a flow control valve contains or that it is not reliable and as it is not they who have to struggle in some dark and inaccessible point in the aircraft in order to remove and replace the defective component, nothing is done to make it more accessible or serviceable.

I do not believe that any component which has to hold a pressure in order to function can ever be either reliable or cheap to produce.

Aircraft would be greatly simplified if the undercarriage, flaps and brakes could all be operated electrically and the compressed air and hydraulic system eliminated altogether. Electric actuators are self-contained and if an actuator becomes defective it can be changed as a unit, whereas the compressed air or hydraulic rams need a maze of pressure accumulators, cut-out valves, pressure-reducing valves, flow control valves, clack valves, selectors and filters and so on.

The complexity and cost of the undercarriage appears to increase with each new design and I have always been amazed that it has been possible to compress so much expense and fine engineering into an apparatus which, essentially, is intended to

perform the same purpose as the wheels of a farm cart, that is, to carry a load across a field. Most designs of undercarriages depend for their functioning on the holding of pressure in the legs which also have to contain a certain fixed amount of fluid. Judging by the aircraft which one sees, the pressure holding part is not very successful as the leg extension is often quite different on aircraft of the same type and between port and starboard legs of the same aircraft.

The undercarriage is only used for a few seconds at the start and finish of each flight; and as the Daily Inspection Schedule most likely calls for the leg pressures to be checked daily it looks as though we do not get much serviceability for all the cost and working to fine limits that has gone into the job. The undercarriage, like the hydraulic system, is usually a proprietary article and can only be repaired or overhauled by fitting replacements which, for aircraft flying on long routes, entails holding stocks of these expensive items at servicing bases along the route.

If aircraft really need all this elaborate equipment for the few seconds during which they are being transferred from the air to the ground it would probably be well to start thinking for the future in terms of sprung runways.

I know of one aircraft handbook which gives instructions for a "quarter of a can" of fluid to be forced into the undercarriage legs. The same book quotes control surface movements in 64ths of an inch, which is no doubt the accurate conversion of the dimension from degrees to chordal measurement, but is not likely to be worked to with a carpenter's rule on a fabric-covered trailing edge.

All servicing information such as pressures, types of fluid and control settings, and so on, should be either in the form of a printed notice permanently fixed in the aircraft or, better still, marked on each component to which it refers.

I have known much delay in the repair and overhaul of aircraft because there was no repair scheme given in the instruction book to cover some damage that had occurred. This entails sending to the parent firm a sketch and description of the damage.

A book of standard repair schemes should be compiled by the A.R.B. which should be applicable to all aircraft of standard types of construction, both British and Foreign, on the lines of A.P. 2662 as used for Service aircraft. This would relieve the aircraft manufacturer of much of the work entailed in producing his own repair manual, and "out of the ordinary" repair schemes could be circulated to the holders of the book in amendment form.

I remember a case where a well-known type of aircraft had to have a defective tube replaced in one of the engine mountings, for the repair of which the maker's repair manual gave the usual repair by insertion of a new length of tube joined to the standing portions by internal liners and welding. Unfortunately, the liner tubes were listed in the repair manual with part numbers for ordering as spares, which caused considerable delay in returning the aircraft to service as the A.R.B. surveyor would not permit the work to proceed until either the drawings of the liners, or their exact length, or the liners themselves, had been obtained from the makers.

It has been general practice for many years that the liners used for this type of tube repair should be of the same gauge and material as the damaged tube and should be twice as long as tube diameter. In another case, a patch repair was required in the fuel tank of a small two-seater built before the war by a firm now non-existent. The tank was made of tinned steel, of which S.20 is the only current specification, but the A.R.B. surveyor would not permit the repair until a piece of the original tank had been removed and subjected to chemical analysis to prove that it also was S.20. The chemical analysis cost as much as a new tank.

These are typical of the sort of standard repair schemes that could be incorporated in the general repair manual advocated above, which could save many weeks in the overhaul time of an aircraft.

W. J. Halland, A.F.R.Ae.S.

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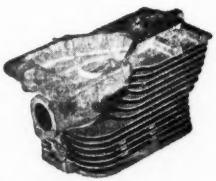

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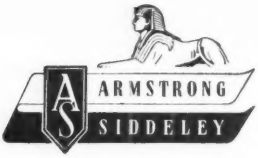

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Williamson Manufacturing Co. Ltd.	Litchfield Gardens, Willesden Green, N.W.10.	Willesden 0073-0075
Yorkshire Engineering Supplies Ltd.	Bronze Foundries, Upper Wortley Road, Leeds, 12	Leeds 38234 & 38291

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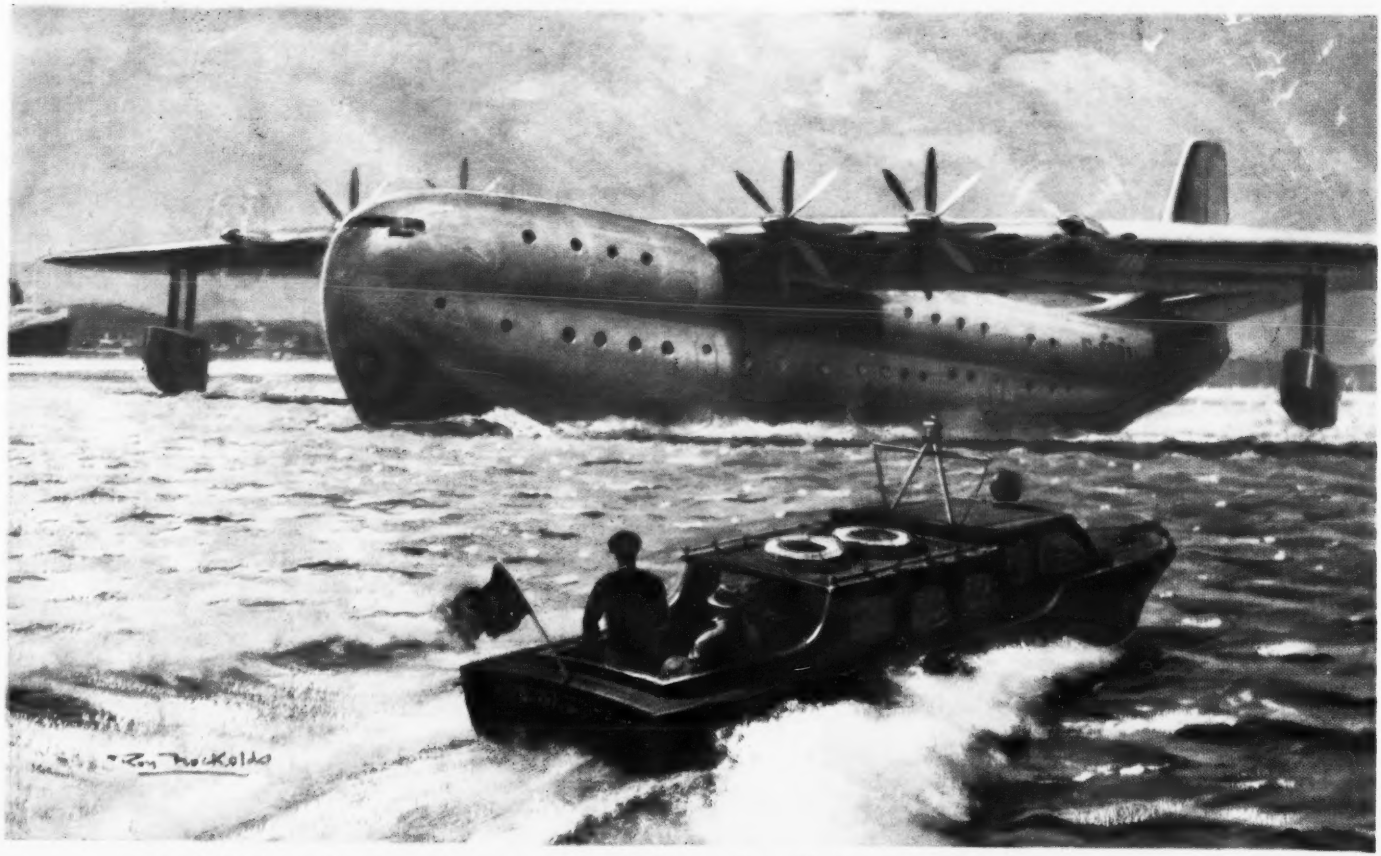

An Artist's impression of one of the Saunders-Roe SR45 140 ton long-range flying boats

SAUNDERS ROE

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